

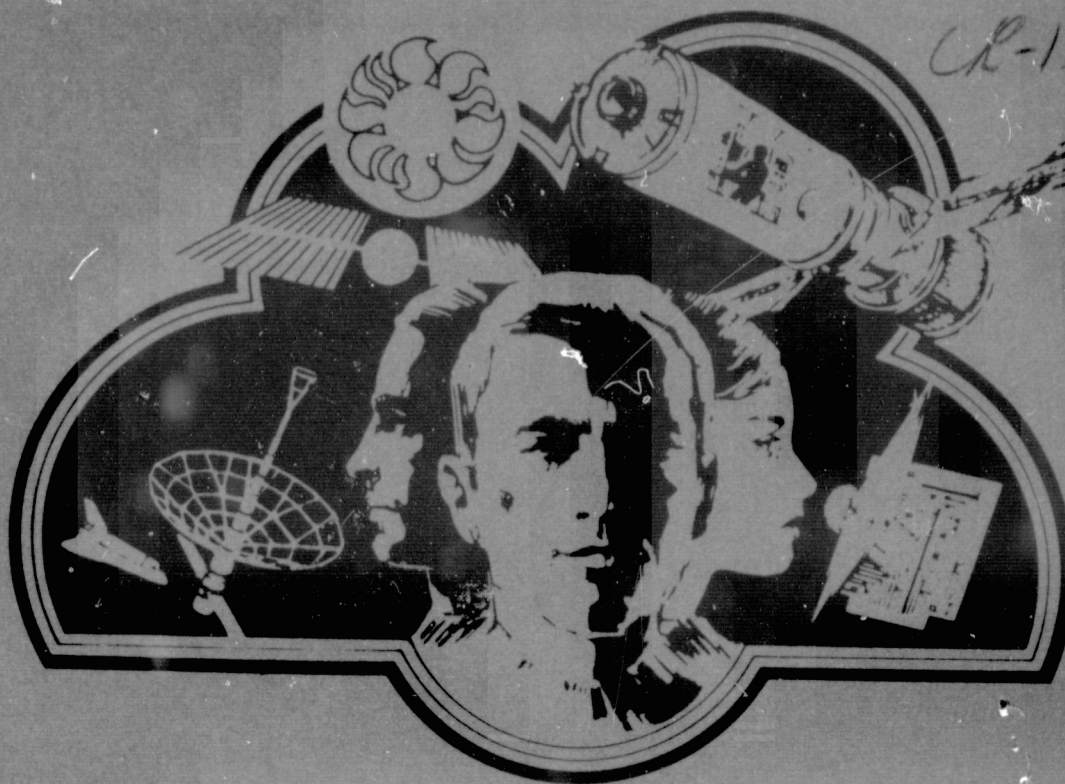
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SPACE STATION SYSTEMS ANALYSIS STUDY PART 3: DOCUMENTATION

VOLUME 3
Appendixes

Book 2 — Supporting Data

(NASA-CR-151501) SPACE STATION SYSTEMS
ANALYSIS STUDY. PART 3: DOCUMENTATION.
VOLUME 3: APPENDIXES. BOOK 2: SUPPORTING
DATA (McDonnell-Douglas Astronautics Co.)
300 p HC A13/MF A01

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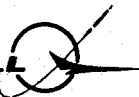
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SPACE STATION SYSTEMS ANALYSIS STUDY

PART 3: DOCUMENTATION

**VOLUME 3
Appendixes
Book 2
Supporting Data**

JULY 1977

MDC G6922

CONTRACT NO. NAS 9-14958
DPD NO. 524
DR NO. MA-04

APPROVED BY:

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PREPARED FOR: NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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PREFACE

The Space Station Systems Analysis Study is a 15-month effort (April 1976 to June 1977) to identify cost-effective Space Station systems options for a manned space facility capable of orderly growth with regard to both function and orbit location. The study activity has been organized into three parts. Part 1 was a 5-month effort to review candidate objectives, define implementation requirements, and evaluate potential program options in low earth orbit and in geosynchronous orbit. Part 2 was also a five-month effort to define and evaluate specific system options within the framework of the potential program options developed in Part 1.

Part 3, the last portion of this study, defines a series of program alternatives and refines associated system design concepts so that they satisfy the requirements of the low earth orbit program option in the most cost-effective manner.

The final reporting of the Part 3 study activity consists of the following:

Volume 1, Executive Summary

Volume 2, Technical Report

Volume 3, Appendixes

Book 1, Supporting Data

Book 2, Supporting Data

Volume 4, Supporting Research and Technology Report

Volume 5, Cost and Schedules Data

A complete list of Parts 1 and 2 tables of contents are included for references in Volume 3, Book 2 in Section 17 of the appendix.

During this study, subcontract support was provided to the McDonnell Douglas Astronautics Company (MDAC) by TRW Systems Group, Aeronutronic Ford Corporation, the Raytheon Company, and Hamilton Standard.

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Section 7

MODULE DEVELOPMENT

7.1 Module Development

The development of the module concepts was predicated upon the realization that the Space Construction Base, whatever its final form, would be composed of a number of functional elements or subsystem assemblies. Major differences in concepts would only arise from the type and quantity of mission hardware proposed to perform the fabrication and/or assembly tasks, the program schedules for task accomplishment, and the degree of support provided by systems such as the Orbiter. Then the identification of the functional elements and their definitions could be employed as building blocks from which all modules would be developed with said development constrained by programmatic considerations.

The first task, as described in Section 5.2, was to identify the functional elements. These elements and the functions performed by each are listed in Table 7-1. The second task, their identification, was preceded by a selection process. This consisted of reviewing what definitions existed due to their preparation in Part 2 or under separate study tasks, whether their importance to the study warranted special handling and/or if their impact on the modules was such as to justify their definition.

As indicated by the lack of a checkmark preceding the element name in the table, no further definition was found to be required for mission related equipment. These items are either covered in Section 3 or have been previously reported on in Part 2 of the study; and, of course, with the exception of the crane, they are external to the modules. For those elements with checks, descriptions were prepared on assembly level data sheets, Figure 7-1 through 7-42.

In preparing these sheets, it was found that the elements either contained large components or were sufficiently complex that they required definition at a lower level than had been anticipated. As a result, elements such as the thermal control system are described on many data sheets. Using these descriptions, the module layouts shown in Section 5.2 were synthesized.

Table 7-1 (Sheet 1 of 3)

PRIMARY FUNCTIONAL ELEMENTS AND FUNCTIONS PERFORMED

Functional Element	Functions
• Crane	• Material/equipment transport • Berthing • Material/equipment positioning and alignment
• Indexing Turntable	• Work rotation • Indexing and alignment
• Strongback	• Equipment/material retention • Material/equipment positioning
• Cherry Picker	• EVA restraint • Equipment/tool storage • Crew/work positioning
• Universal Truss Assembly Jig	• Equipment/material retention • Material/equipment positioning • Beam and truss assembly
• Truss Assembly Jig (4M)	• Truss assembly
• Solar Collector F/A Jig	• Equipment/material retention • Roll-forming machine mounts • Reflector rolls and solar cell roll mounts
• Metal Roll Forming Unit	• Solar array assembly • Beam fabrication • Truss assembly
• Composites Fabrication Unit	• ~1.0-m-dia composite tube fabrication
• Composites 20-cm Tube Fabrication Unit	• 20-cm-dia composite tube fabrication
• Composites Tube Weaving Fab Unit	• Beam cap fabrication • Truss beam fabrication • 1-m beam assembly
• Truss Support Tool	• Transfer of objective element to within crane reach

Table 7-1 (Sheet 2 of 3)

PRIMARY FUNCTIONAL ELEMENTS AND FUNCTIONS PERFORMED

Functional Element	Functions
• Satellite	• Antenna beam mapping • RFI mapping • Signal source simulation
• Assembly Beam	• Equipment/material support-retention • Equipment/material positioning
• Telescoping Pallet	• Equipment/material support-retention • Equipment/material positioning • (part of structure)
• Power System	• Power generation • Energy storage • Switching and distribution
• EVA Airlock	• Pressurization and depressurization • Atmosphere storage
• Crew Quarters	• Restrains crew during sleep periods • Personal effects storage • Privacy control
• SCB Control Station	• Operations management • Status display and equipment control • Resource management
• Reaction Control Modules	• Fuel storage and transfer • Attitude control and station keeping
• Electronics Racks	• Communications signal processing • Ephemeris calculation and control • Sensor data processing
• Docking/Berthing Ports	• Orbiter/module docking/berthing • Utility connection • Crew/equipment ingress and egress

Table 7-1 (Sheet 3 of 3)

PRIMARY FUNCTIONAL ELEMENTS AND FUNCTIONS PERFORMED

Functional Element	Functions
● Extravehicular Mobility Unit and Maintenance Area	● Suit drying and suit/EMU storage
	● Suit/EMU maintenance and resupply
	● Personal rescue unit storage
● Environmental Control and Life Support Assemblies	● Consumables storage
	● Atmosphere control
	● Atmosphere constituent monitoring
● Thermal Control Assemblies	● Heat rejection
	● Temperature control
	● Coolant loop control
● Waste Management Assemblies	● Personal hygiene
	● Crew waste collection
● View Port/Window Assemblies	● Equipment observation
	● EVA observation
	● Earth/space observation
● Food Management	● Galley
	● Food storage (pantry)
	● Dining area
● Logistics Volume	● Spares storage
	● Consumables storage
● IMU/Horizon Sensor Assembly	● Attitude sensing
	● Rate sensing
● Recreation Area	● Recreation
	● Exercise
● Dispensary	● Medical care
	● Medical supply storage
● Test Control Station	● Objective element checkout
	● Component checkout
	● Operations supervision
● MMU Station	● MMU storage
	● MMU maintenance
● Ku Band Antenna Assembly	● High data rate transmission and reception

1. NAME: ECLS (POTABLE WATER STORAGE)

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

24.1

8

1

0

VOL. (M³)

.23

.08

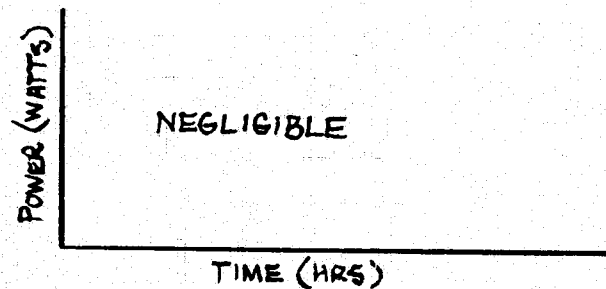
NEGL.

0

3. SUPPORT REQUIREMENTS

- o 1/4 INCH LINE TO WATER RECOVERY ASSY
- o WATER LINE TO POTABLE WATER RESUPPLY ASSY (LOGISTICS MODULE)
- o CDMS FOR QUANTITY AND PRESSURE

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: NEAR WATER
RECOVERY ASSY
ACCESS: ACCESS REQUIRED
ON SIDES A AND B

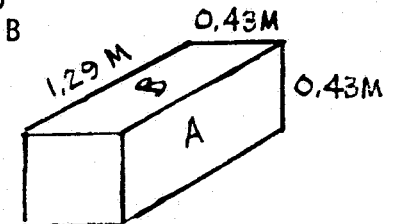


Figure 7-1. Assembly Level Data Sheet — ECLS (Potable Water Storage)

1. NAME: ECLS (POTABLE WATER RECOVERY)

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

VOL. (M³)

305

1.56

150

0.07

10

.1

174

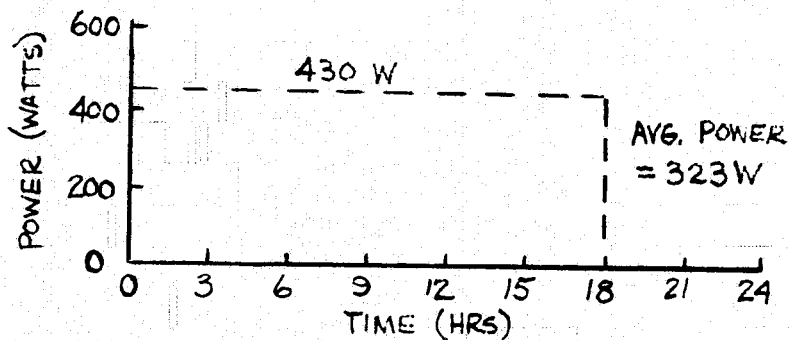
1.43

3. SUPPORT REQUIREMENTS

- o AVERAGE AIR COOLING OF 301 WATTS
- o CDMS FOR CHECKOUT AND MONITORING
- o 30 PSIA N₂ SUPPLY
- o 1/4 INCH LINE TO WATER STORAGE ASSY

4. POWER PROFILE

5. EXTERNAL CHARACTERISTICS



LOCATION: NEAR WASTE WATER SOURCES
AND POTABLE WATER STORAGE
ACCESS: ALL SIDES EXCEPT B

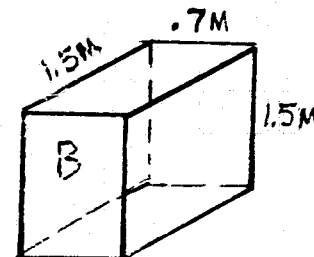


Figure 7-2. Assembly Level Data Sheet — ECLS (Potable Water Recovery)

1. NAME: ECLS (ATMOSPHERE RECONDITIONING)

NO. REQUIRED: 1/MODULE

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

VOL. (M³)

163

.5

70

.2

25

.1

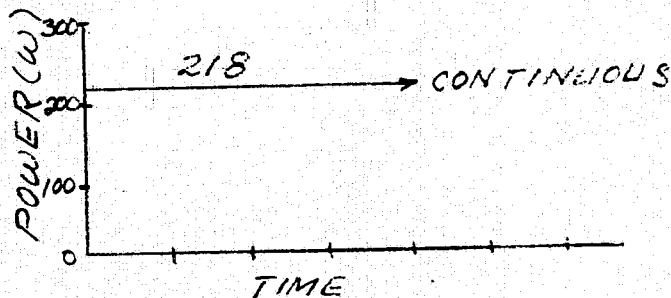
0

0

3. SUPPORT REQUIREMENTS

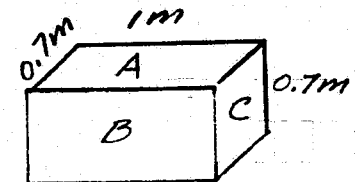
- o HEAT REJECTION REQUIRED BY CO₂ REMOVAL = 56W
- o 865 KG/HR COOLING WATER
- o CDMS FOR CHECKOUT AND MONITORING

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: CENTRAL LOCATION
IN MODULE - NEAR OVERBOARD DUCT



ACCESS: REQUIRED ON SIDES A & B,
SIDE C MUST NOT BE OBSTRUCTED FOR
AIR FLOW

Figure 7-3. Assembly Level Data Sheet — ECLS (Atmosphere Reconditioning)

1. NAME: ECLS (PRESSURE CONTROL)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

NO. REQUIRED: 1/PRESSURE COMPARTMENT

WT. (KG)

VOL. (M³)

8

.06

6

.04

3.6

.02

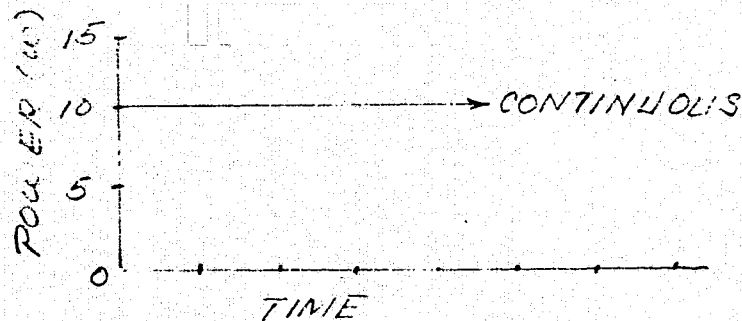
0

0

3. SUPPORT REQUIREMENTS

- o CDMS FOR CHECKOUT AND MONITORING

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: MAIN ECLSS
PANEL (CONSOLE)

ACCESS: TWO-SIDED ACCESS
REQUIRED

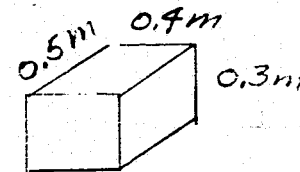


Figure 7-4. Assembly Level Data Sheet - ECLS (Pressure Control)

1. NAME: ECLS (REPRESSURIZATION GAS)

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES (PER REPRESSURIZATION)

WT. (KG)

115

2

.5

91

VOL. (M³)

.8

NEGL.

NEGL.

.8

3. SUPPORT REQUIREMENTS

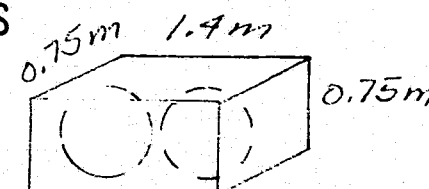
- o DATA MANAGEMENT FOR QUANTITY, PRESSURE AND TEMPERATURE.

4. POWER PROFILE

NONE

5. EXTERNAL CHARACTERISTICS

LOCATION: CS OR CSM



ACCESS: MUST BE ABLE TO
REPLACE TANKS WHEN DEPLETED.

Figure 7-5. Assembly Level Data Sheet — ECLS (Prepressurization Gas)



1. NAME: ACLS (MAKEUP O₂ AND N₂) STORAGE

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY) TANKS FOR 180 DAYS
- o ONBORAD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES (180 DAYS)

WT. (KG)

1740

2

.5

1358

VOL. (M³)

5.2

NEGL.

NEGL.

5.2

3. SUPPORT REQUIREMENTS

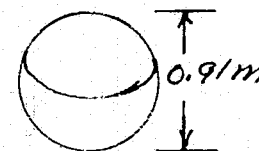
- o DATA MANAGEMENT FOR QUANTITY, PRESSURE AND
AND TEMPERATURE

4. POWER PROFILE

NEGLECTIBLE

5. EXTERNAL CHARACTERISTICS

LOCATION: LOGISTICS MODULE



ACCESS: REQUIRED FOR FILL LINE
AND VALVE REPLACEMENT

Figure 7-6. Assembly Level Data Sheet — ECLS (Makeup O₂ and N₂ Storage)

1. NAME: ECLS (DUMP AND RELIEF VALVES)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

NO. REQUIRED: 1/PRESSURIZED COMPARTMENT

WT. (KG)

VOL. (M³)

7

.04

5

.03

1

.01

0

0

3. SUPPORT REQUIREMENTS

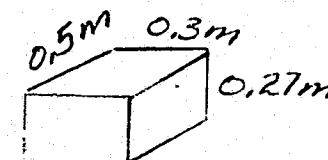
- o CDMS FOR CHECKOUT AND MONITORING

4. POWER PROFILE

NONE

5. EXTERNAL CHARACTERISTICS

LOCATION: ON OVERBOARD DUCT



ACCESS: THREE SIDES REQUIRED.

Figure 7-7. Assembly Level Data Sheet — ECLS (Dump and Relief Valves)

1. NAME: ECLS (POTABLE WATER RESUPPLY ASSEMBLY)

NO. REQUIRED: 1/LOGISTICS MODULE

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

VOL. (M³)

730

10.3

3

.01

0

0

4760

4.8

3. SUPPORT REQUIREMENTS

- o 1/4 INCH LINE TO WATER STORAGE UNIT
- o CDMS FOR PRESSURE AND QUANTITY
- o 100 PSIA N₂ SUPPLY

4. POWER PROFILE

NEGLIGIBLE

5. EXTERNAL CHARACTERISTICS

LOCATION: LOGISTICS MODULE

ACCESS: REQUIRED TO VALVES

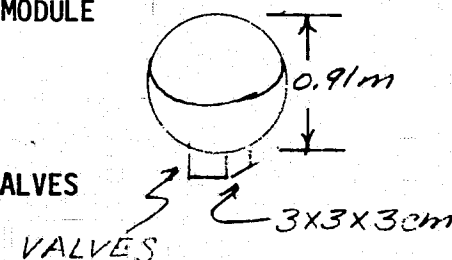


Figure 7-8. Assembly Level Data Sheet — ECLS (Potable Water Resupply Assembly)

1. NAME: ECLS (FIRE DETECTION AND SUPPRESSION)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)*
- o RESUPPLY SPARES (180 DAYS)*
- o RESUPPLY EXPENDABLES (180 DAYS)*

*REQUIRED FOR EIGHT INSTALLED ASSEMBLIES

NO. REQUIRED: 1/EQUIPMENT RACK

WT. (KG)

3.7

7.4

2.0

2.7

VOL. (M³)

.002

.004

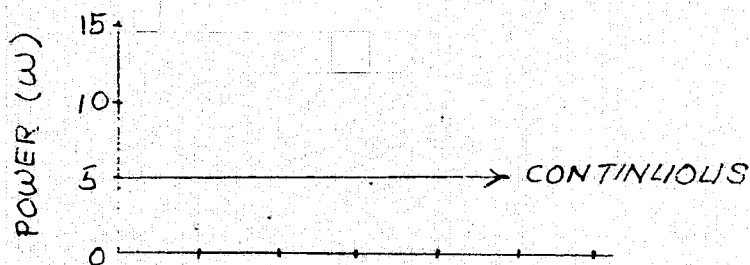
.001

.001

3. SUPPORT REQUIREMENTS

- o CEMS FOR FIRE ALARM AND CHECKOUT

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: ONE INSTALLED IN EACH CONSOLE WITH POTENTIAL FIRE HAZARD.

ACCESS: REQUIRED TO REPLACE CO₂ BOTTLE AND SMOKE DETECTOR UNIT.

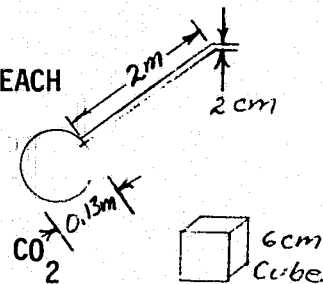


Figure 7-9. Assembly-Level Data Sheet - ECLS (Fire Detection and Suppression)

1. NAME: ECLS (180-HOUR EMERGENCY PALLET)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES*

*REFURBISHED AFTER USE

3. SUPPORT REQUIREMENTS

- o CDMS FOR STATUS MONITORING
- o OVERBOARD DUCT FOR WATER EVAPORATOR

4. POWER PROFILE

NEGLIGIBLE

NO. REQUIRED: 1/REFUGE

WT. (KG)

698

0

0

0

VOL. (M³)

1.45

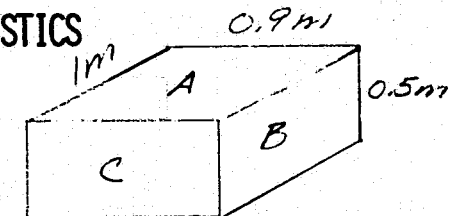
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5. EXTERNAL CHARACTERISTICS

LOCATION:



ACCESS: REQUIRED ON SIDES

A, B, AND C

Figure 7-10. Assembly Level Data Sheet - ECLS (180-Hour Emergency Pallet)

1. NAME: ECLS (AIRLOCK PUMPDOWN)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

NO. REQUIRED: 1

WT. (KG)

15.3

8

.5

0

VOL. (M³)

.06

.03

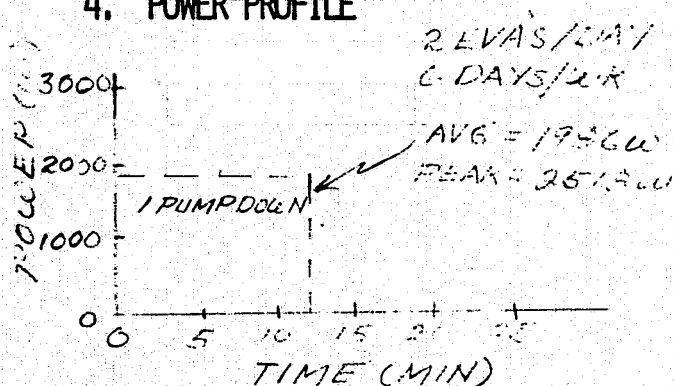
NEGL.

0

3. SUPPORT REQUIREMENTS

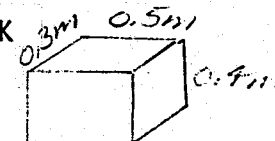
- o CDMS FOR CHECKOUT AND MONITORING
- o AC POWER

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: ADJACENT TO AIRLOCK



ACCESS: THREE SIDES REQUIRED

Figure 7-11. Assembly Level Data Sheet — ECLS (Airlock Pumpdown)

1. NAME: IMU/HORIZON SENSOR ASSEMBLY

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

3. SUPPORT REQUIREMENTS

- o HEAT REJECTION

NO. REQUIRED: 1

WT. (KG)

25.85 INSIDE

16.33 OUTSIDE

VOL. (M³)

.0387 INSIDE

.0151 OUTSIDE

MINOR (SEE POWER PROFILE)

4. POWER PROFILE

113W CONTINUOUS

5. EXTERNAL CHARACTERISTICS

LOCATION:

ACCESS:

SEE STTACHED SKETCH

Figure 7-12. Assembly-Level Data Sheet — IMU/Horizon Sensor Assembly

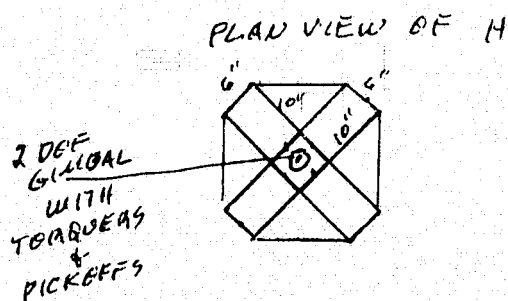
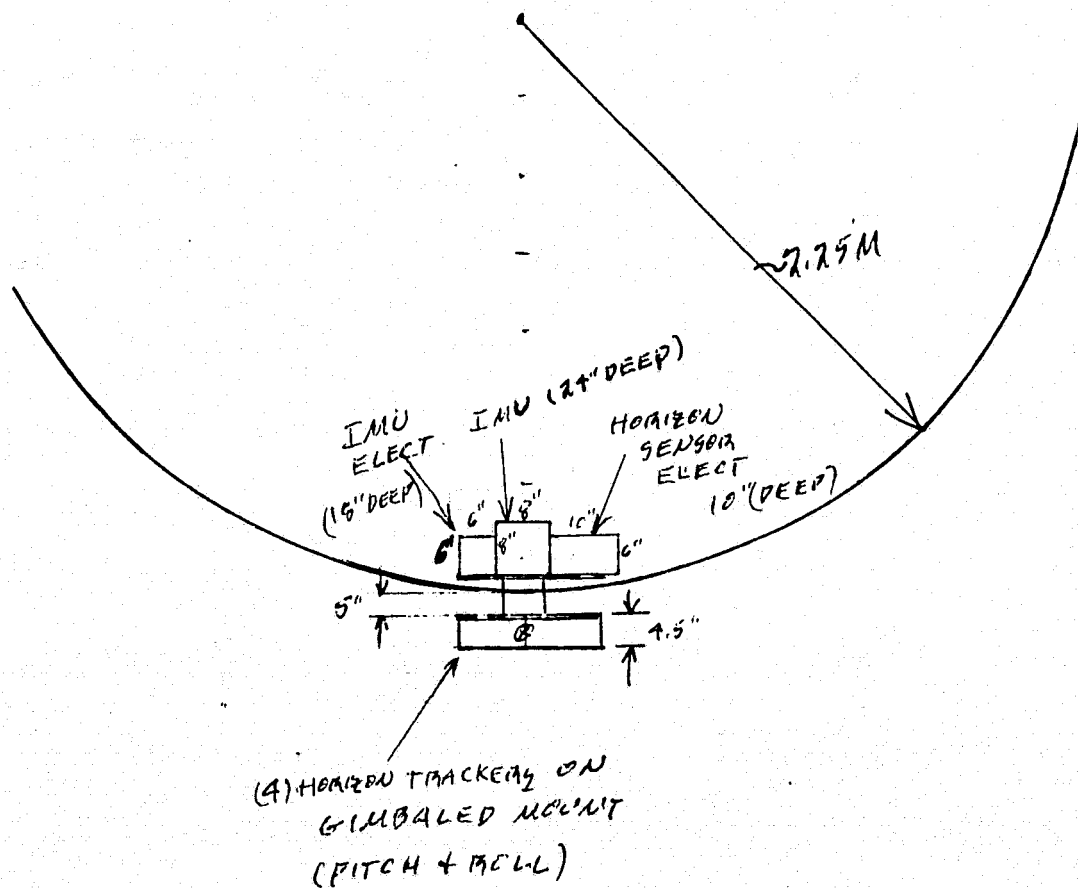


Figure 7-13. IMU/Horizon Sensor Installation

1. NAME: REACTION CONTROL MODULES
(PROPULSION SYSTEM PALLET)

NO. REQUIRED: 4

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

290

N/A

N/A

2326

VOL. (M³)

2.3*

N/A

N/A

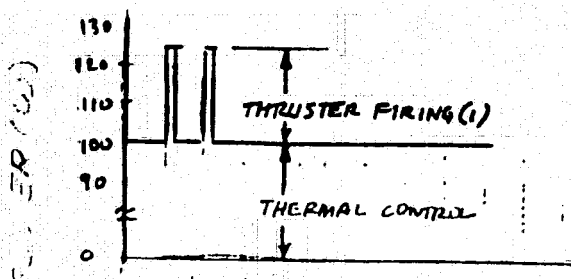
2.3*

*PROPELLANT AND PRESSURIZATION BOTTLES ONLY, 90-DAY SUPPLY

3. SUPPORT REQUIREMENTS

- o FUEL RESUPPLY

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: END OF CS OR CSM

ACCESS: N/A

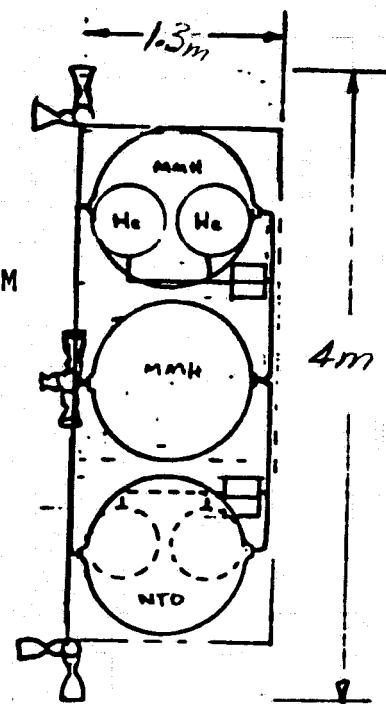


Figure 7-14. Assembly Level Data Sheet — Reaction Control Modules (Propulsion System Pallet)

1. NAME: EXTRAVEHICULAR MOBILITY UNIT

NO. REQUIRED: (4 SUITS)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES***

WT. (KG)

VOL. (M³)

434*

2.12**

TBD

TBD

O₂ 693

H₂O 2558

2.18

1533

CCC & BATTERIES

*INCLUDES WEIGHT OF 4 EMUs

**INCLUDES 1 RECHARGE STATION AND 4 DONN/SUIT STORAGE STATIONS

***ASSUMES 4 EVAs/DAY, 6 DAYS A WEEK

3. SUPPORT REQUIREMENTS

- o HEAT REJECTION (BTU/HR)
- o DATA STORAGE (WORDS)
- o AREA ILLUMINATION (LUM/M²)
- o VIDEO BW (MHz)/DATA RATE (MBPS)

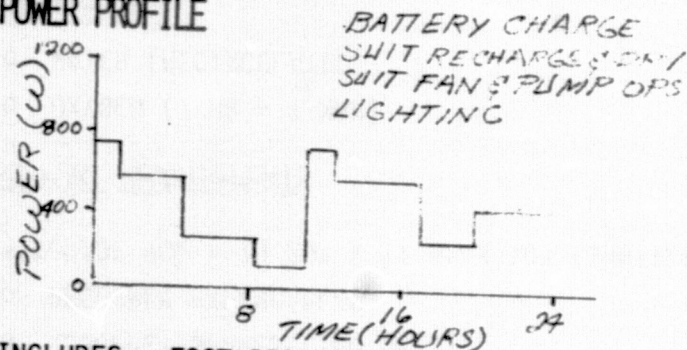
TBD

NONE

110

o

4. POWER PROFILE



INCLUDES: FOOT RESTRAINTS
HAND RAILS
SUIT VOLUME

5. EXTERNAL CHARACTERISTICS

LOCATION: ADJACENT TO EVA AIRLOCK

INCLUDES:

FOOT RESTRAINTS

HAND RAILS

SUIT VOLUME

ACCESS:

Figure 7-15. Assembly Level Data Sheet — Extravehicular Mobility Unit

1. NAME: PERSONNEL RESCUE UNIT

NO. REQUIRED: 4

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

4.6

N/A

N/A

N/A

VOL. (M³)

* (18 FT³)

N/A

N/A

N/A

*STORAGE VOL = 19.25" X 17.5" X 10" (SHUTTLE STOWAGE LOCKER)

3. SUPPORT REQUIREMENTS

- o OXYGEN (7 HR - 1 MAN)
- o WATER (RECIRCULATED)

CONTAINS 1 HR SUPPLY

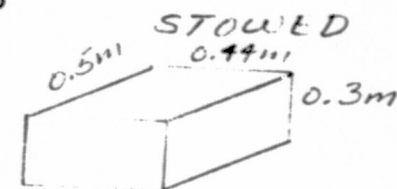
4. POWER PROFILE

NONE REQUIRED

5. EXTERNAL CHARACTERISTICS

LOCATION: INTERNAL

ACCESS: ZIPPER OPENING



UNSTOWED

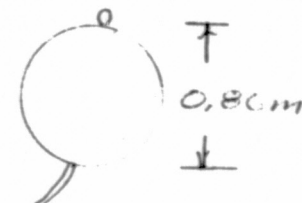


Figure 7-16. Assembly Level Data Sheet -- Personnel Rescue Unit

1. NAME: ELECTRONICS RACKS

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

3. SUPPORT REQUIREMENTS

- o COLD PLATES (We)
- o POWER
- o CHECKOUT AND MAINTENANCE (HR/DAY)

NO. REQUIRED: 1 SET (2 RACKS)

WT. (KG)

VOL. (M³)

545

1.7

110

0.3

30

0.1

34

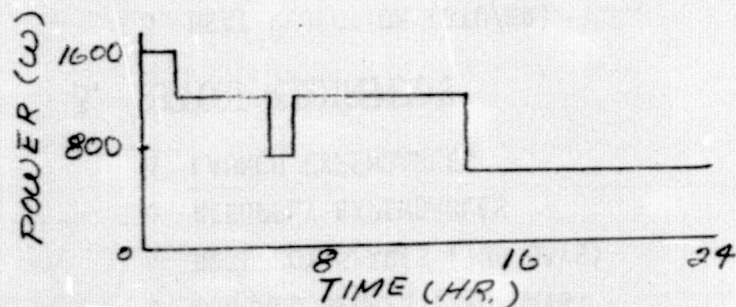
0.15

1000

(SEE PROFILE)

1

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: INTERNAL - 2 REQUIRED

ACCESS: FRONT - SLIDE MOUNTS

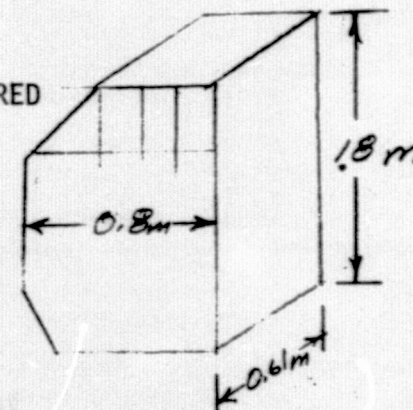


Figure 7-17. Assembly Level Data Sheet - Electronics Racks

1. NAME: WASTE MANAGEMENT (PERSONAL HYGIENE)

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES
- o LAUNCH EXPENDABLES

WT. (KG)

VOL. (M³)

19.9

0.14

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3. SUPPORT REQUIREMENTS

- o HEAT REJECTION (BTU/HR)
- o DATA STORAGE (WORDS)
- o AREA ILLUMINATION (LUM/M²)
- o VIDEO BW (MHz)/DATA RATE (MBPS)

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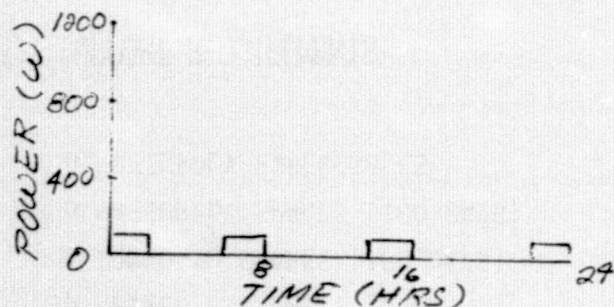
300-500

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--

--

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: CS NEAR TOILET/URINAL

ACCESS: FRONT ACCESS FOR UTILIZATION AND MAINTENANCE

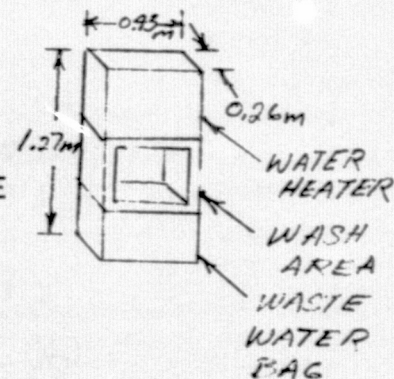


Figure 7-18. Assembly Level Data Sheet — Waste Management (Personnel Hygiene)

1. NAME: WASTE MANAGEMENT (CREW WASTE COLLECTION) NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES
- o LAUNCH EXPENDABLES

WT. (KG)

VOL. (M³)

--	0.31
--	--
--	--
--	--
--	--
--	1.84

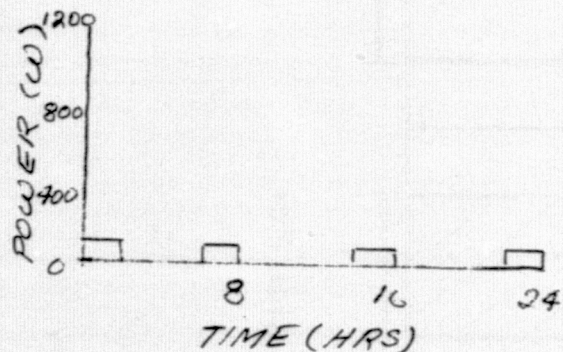
3. SUPPORT REQUIREMENTS

- o HEAT REJECTION (BTU/HR)
- o DATA STORAGE (WORDS)
- o AREA ILLUMINATION (LUM/M²)
- o VIDEO BW (MHz)/DATA RATE (MBPS)

300-500

--

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: PERSONAL HYGIENE COMPARTMENT

ACCESS: TOP AND SIDE ACCESS REQUIRED

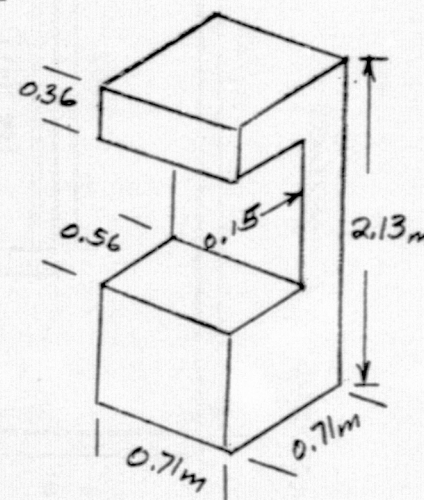


Figure 7-19. Assembly Level Data Sheet — Waste Management (Crew Waste Collection)

DIMENSIONS IN METERS

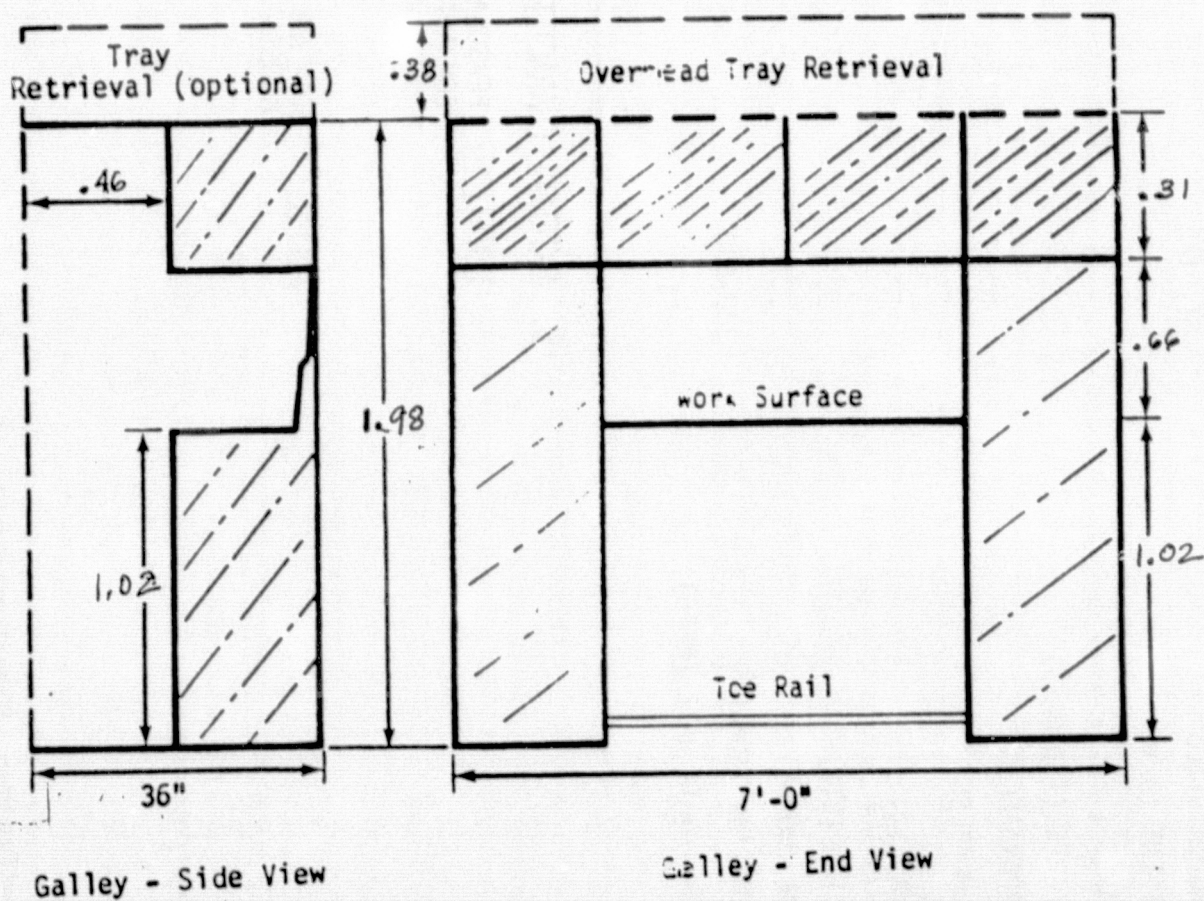
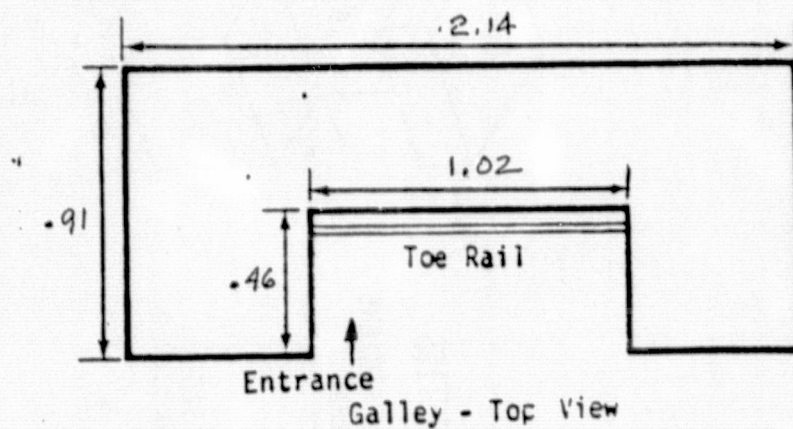


Figure 7-20. Galley

1. NAME: FOOD MANAGEMENT (GALLEY UNIT)

NO. REQUIRED:

2. PHYSICAL CHARACTERISTICS

WT. (KG)

VOL. (M³)

o FIXED EQUIPMENT (DRY)

185.1*

1.16 (GALLEY UNIT)

o ONBOARD SPARES (180 DAYS)

--

--

o RESUPPLY SPARES (180 DAYS)

--

--

o RESUPPLY EXPENDABLES

--

--

o LAUNCH EXPENDABLES

--

--

*WEIGHT DOES NOT INCLUDE WEIGHT FOR STRUCTURE OR STOWED FOOD.

3. SUPPORT REQUIREMENTS

o HEAT REJECTION (BTU/HR)

--

--

o DATA STORAGE (WORDS)

--

--

o AREA ILLUMINATION (LUM/M²)

300-500

--

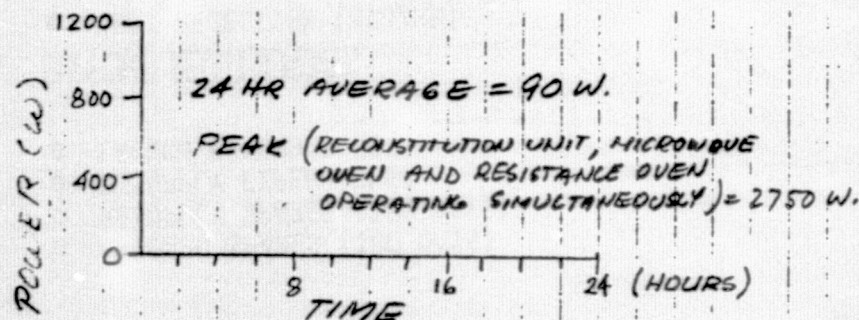
o VIDEO BW (MHz)/DATA RATE (MBPS)

--

--

4. POWER PROFILE

5. EXTERNAL CHARACTERISTICS



LOCATION: GALLEY COMPARTMENT

SEE ATTACHMENT

ACCESS: FRONT ACCESS FOR UTILIZATION REQUIRED. SIDE ACCESS FOR MAINTENANCE REQUIRED FOR MAINTAINABLE EQUIPMENT.

Figure 7-21. Assembly Level Data Sheet — Food Management (Galley Unit)

1. NAME: FOOD MANAGEMENT (DINING AREA)

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

o FIXED EQUIPMENT (DRY)

WT. (KG)

VOL. (M³)

TABLE DIMENSIONS

0.76M HIGH
0.91M WIDE
1.52M LONG

- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES
- o LAUNCH EXPENDABLES

--
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--
--

--
--
--
--

3. SUPPORT REQUIREMENTS

- o HEAT REJECTION (BTU/HR)
- o DATA STORAGE (WORDS)
- o AREA ILLUMINATION (LUM/M²)
- o VIDEO BW (MHz)/DATA RATE (MBPS)

--
--
300-500
--

--
--
--
--

4. POWER PROFILE

NONE

5. EXTERNAL CHARACTERISTICS

LOCATION: DINING/RECREATION AREA

SEE SHEET ATTACHED
TO SEATING RESTRAINTS/CHAIRS

ACCESS: FOUR-SIDED ACCESS FOR
SEATING AROUND TABLE

Figure 7-22. Assembly Level Data Sheet — Food Management (Dining Area)

1. NAME: FOOD MANAGEMENT (SEATING RESTRAINTS/
CHAIRS)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES
- o LAUNCH EXPENDABLES

3. SUPPORT REQUIREMENTS

- o HEAT REJECTION (BTU/HR)
- o DATA STORAGE (WORDS)
- o AREA ILLUMINATION (LUM/M²)
- o VIDEO BW (MHz)/DATA RATE (MBPS)

4. POWER PROFILE

NONE

NO. REQUIRED: 7

WT. (KG)

VOL. (M³)

SEAT DIMENSIONS: 0.31 X 0.46 X 0.61M (HIGH)

300-500

5. EXTERNAL CHARACTERISTICS

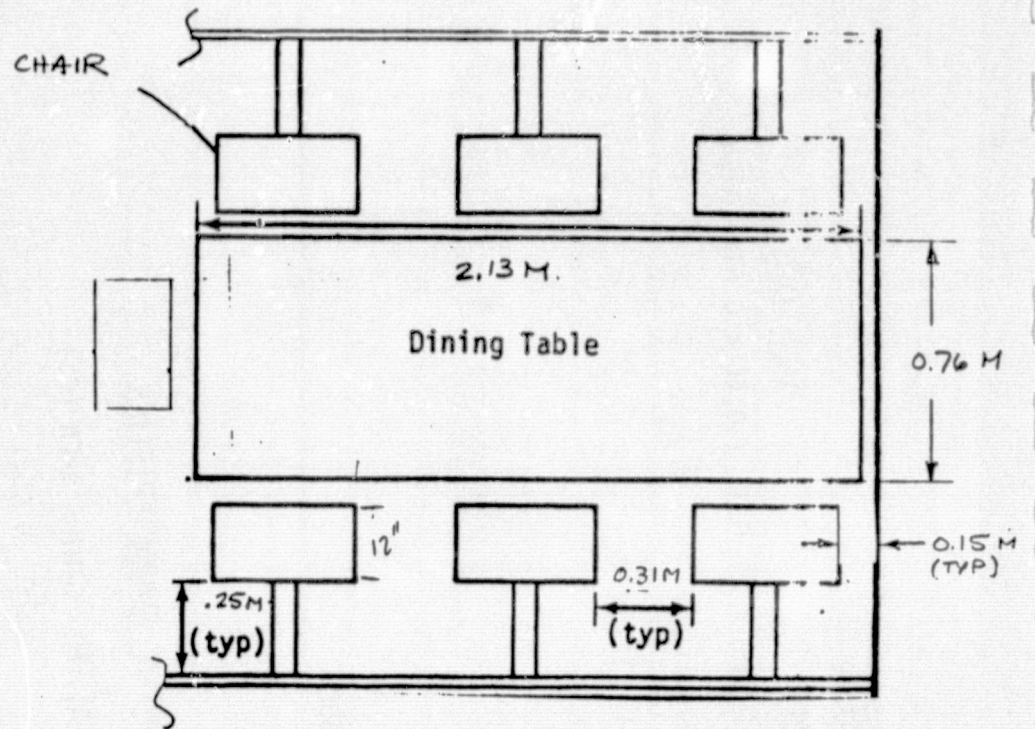
LOCATION: DINING AREA (CS)

SEE ATTACHMENT

ACCESS: MINIMUM DIST. BETWEEN
SEATS = 0.31M

Figure 7-23. Assembly Level Data Sheet — Food Management (Seating Restraints/Chairs)

TOP VIEW



SIDE VIEW

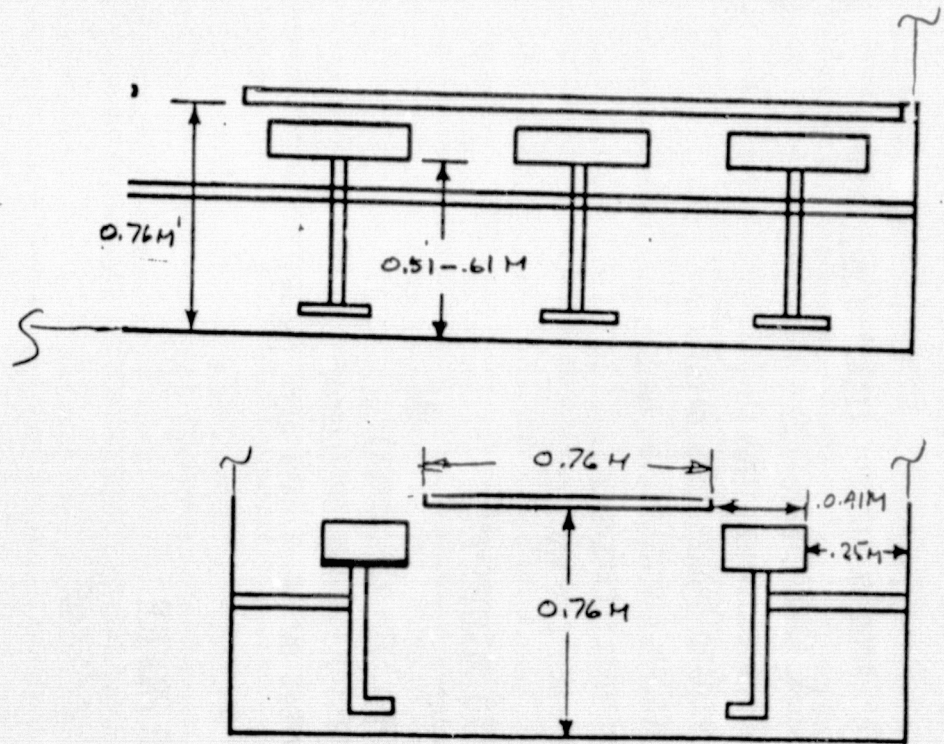


Figure 7-24. Seating Arrangement

1. NAME: FOOD MANAGEMENT (GALLEY - INVENTORY CONTROL UNIT)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES
- o LAUNCH EXPENDABLES

NO. REQUIRED: 1

WT. (KG)

18.14

VOL. (M³)

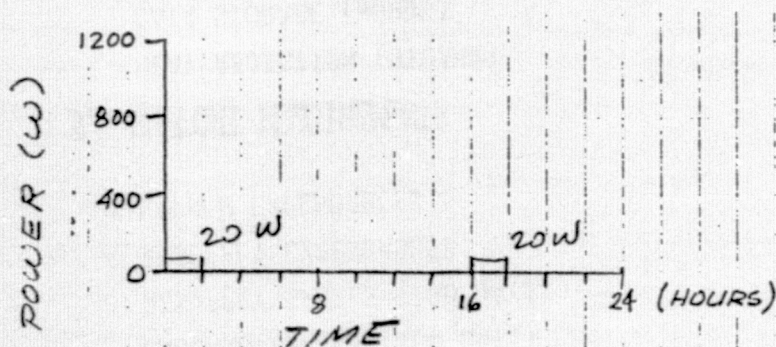
0.04

3. SUPPORT REQUIREMENTS

- o HEAT REJECTION (BTU/HR)
- o DATA STORAGE (WORDS)
- o AREA ILLUMINATION (LUM/M²)
- o VIDEO BW (MHz)/DATA RATE (MBPS)

300-500

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: GALLEY/DINING AREA

ACCESS: FRONT ACCESS REQUIRED

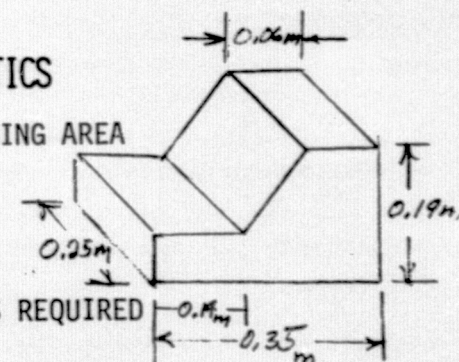


Figure 7-25. Assembly Level Data Sheet — Food Management (Galley-Inventory Control Unit)

1. NAME: CREW QUARTERS (SLEEPING RESTRAINTS/BUNKS) NO. REQUIRED: 7

2. PHYSICAL CHARACTERISTICS

FIXED EQUIPMENT DRY

ONBOARD SPARES (180 DAYS)

RESUPPLY SPARES (180 DAYS)

RESUPPLY EXPENDABLES

LAUNCH EXPENDABLES

WT. (KG)

9.1

VOL. (M³)

3. SUPPORT REQUIREMENTS

HEAT REJECTION (BTU/HR)

DATA STORAGE (WORDS)

AREA ILLUMINATION (LUM/M²)

VIDEO BW (MHz)/DATA RATE (MBPS)

0-500

4. POWER PROFILE

NONE

5. EXTERNAL CHARACTERISTICS

LOCATION: CREW QUARTERS (BUNK)

SEE ATTACHMENT

ACCESS: FRONT OR SIDE ACCESS
DEPENDENT ON ORIENTATION

Figure 7-26. Assembly Level Data Sheet — Crew Quarters (Sleeping Restraints/Bunks)

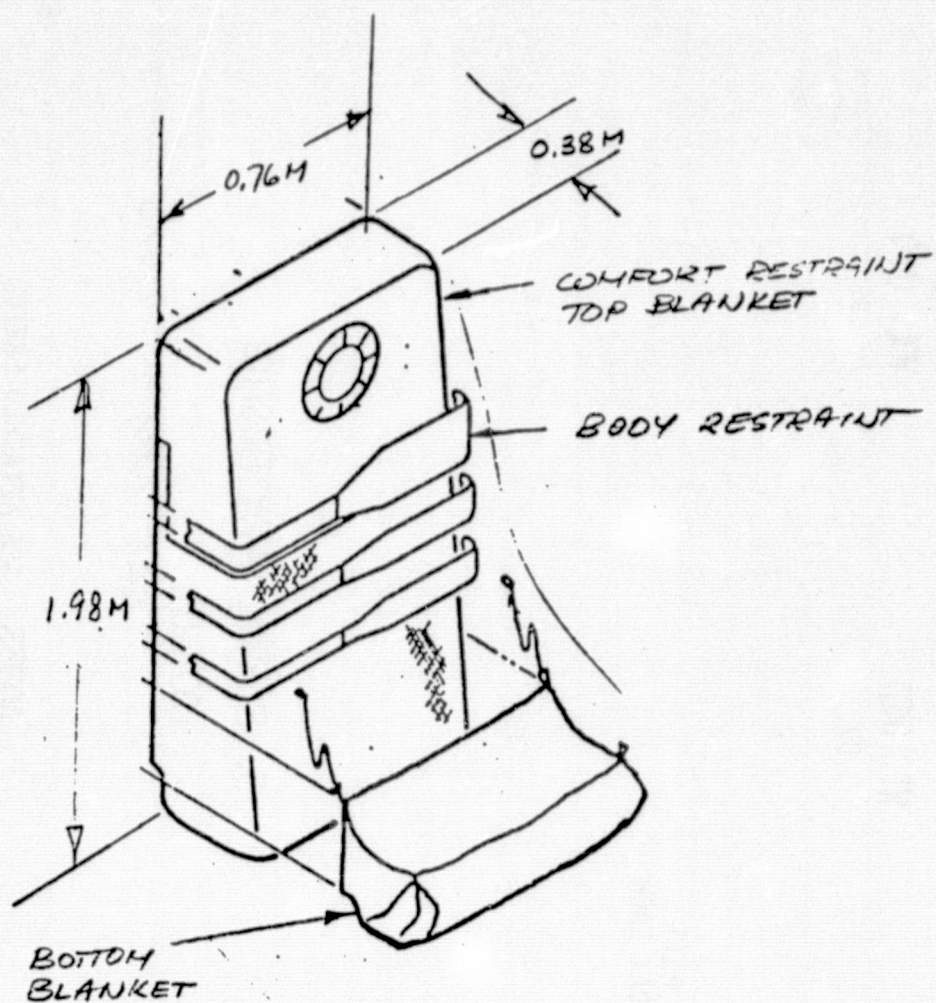


Figure 7-27. Sleep Restraint/Bunk

1. NAME: MANNED MANEUVERING UNIT

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

WT. (KG)

VOL. (M³)

- o FIXED EQUIPMENT (DRY) (MMU)
 - o ONBOARD SPARES (180 DAYS)
 - o RESUPPLY SPARES (180 DAYS)
 - o RESUPPLY EXPENDABLES*
GN₂ 11M/SEC/TK @ 211 KG/CM² (3000 PSI)
ASSUME TWO UNITS USED TWO HR/DAY EACH
- *SIZED TO SUPPORT 6.5 HRS EVA

89

.93

3. SUPPORT REQUIREMENTS

- o FLIGHT SUPPORT STATION
- o CS CAUTION AND WARNING

38

3.5

4. POWER PROFILE

NEGLECTIBLE

5. EXTERNAL CHARACTERISTICS

LOCATION:

SEE ATTACHMENTS

ACCESS: SEE ATTACHED SKETCH

Figure 7-28. Assembly Level Data Sheet — Manned Maneuvering Unit

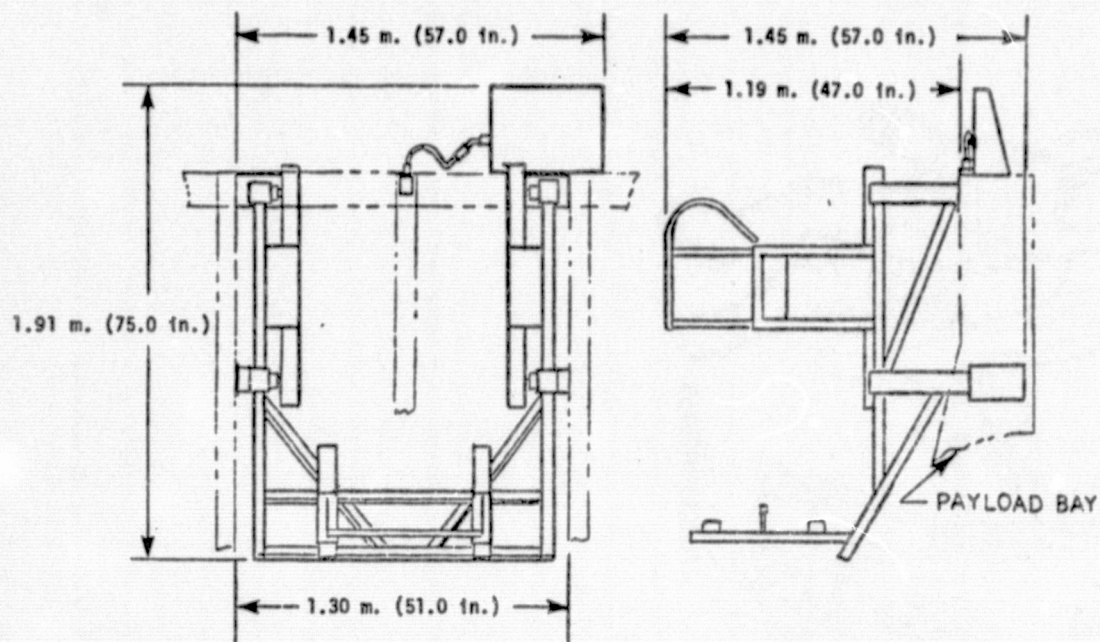


Figure 7-29. Flight Support System Overall Dimensions (Preliminary)

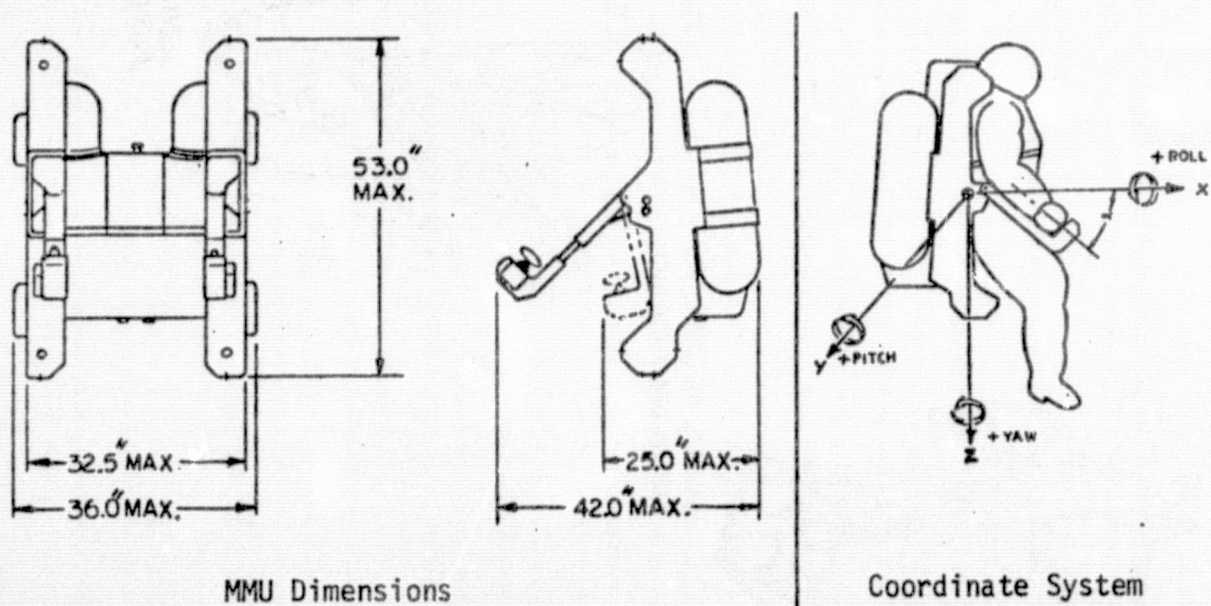
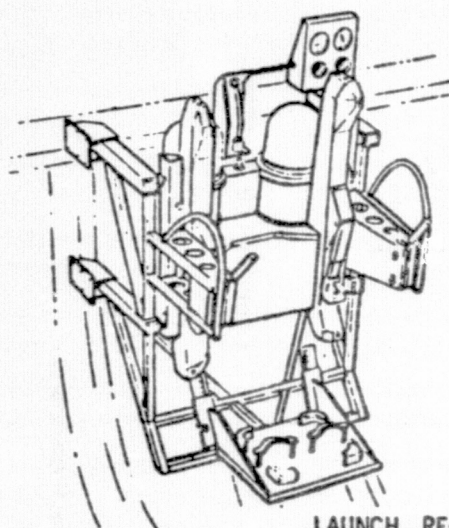
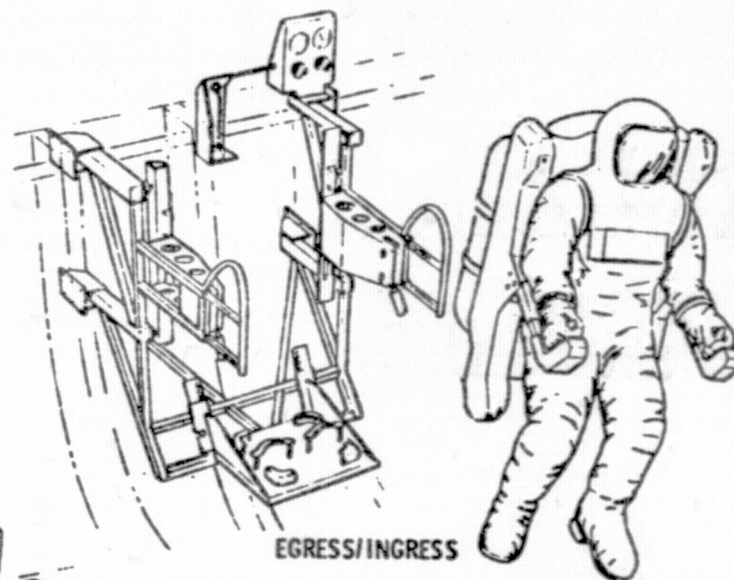


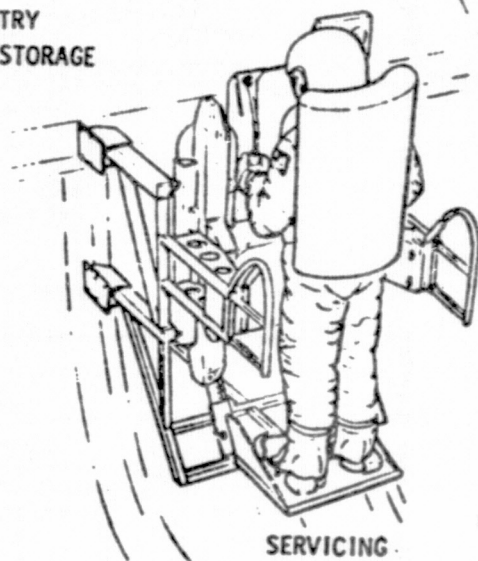
Figure 7-30. MMU Overall Dimensions and Reference Coordinates (Preliminary)



LAUNCH, RE-ENTRY
AND ON-ORBIT STORAGE



EGRESS/INGRESS



SERVICING
(PROPELLANT CHARGE)

Figure 7-31. Manned Maneuvering Unit Flight Support Station

1. NAME: SCB CONTROL STATION

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY):
- o ONBOARD SPARES (180 DAYS):
- o RESUPPLY SPARES (180 DAYS):
- o RESUPPLY EXPENDABLES:

NO. REQUIRED: 1

WT. (KG)

368

74

4

32

VOL. (M³)

1.2

0.24

0.06

0.1

3. SUPPORT REQUIREMENTS

- o HEAT REJECTION (BTU/HR):
- o DATA STORAGE (WORDS):
- o AREA ILLUMINATION (LUM/M²):
- o VIDEO BW (MHz)/DATA RATE (MBPS):

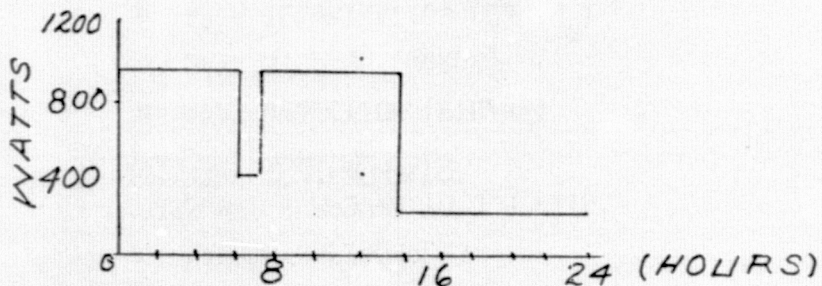
2.07×10^3

8×10^3

100 (MAX)

3.4/1

4. POWER PROFILE



TYPE

REG. DC: 28^{+3}_{-2} V

UNREG. DC: N.A.

OTHER: 5, 15 VDC

5. EXTERNAL CHARACTERISTICS

LOCATION: INTERNAL

ACCESS: HINGE CONSOLE,
REMOVABLE PANELS

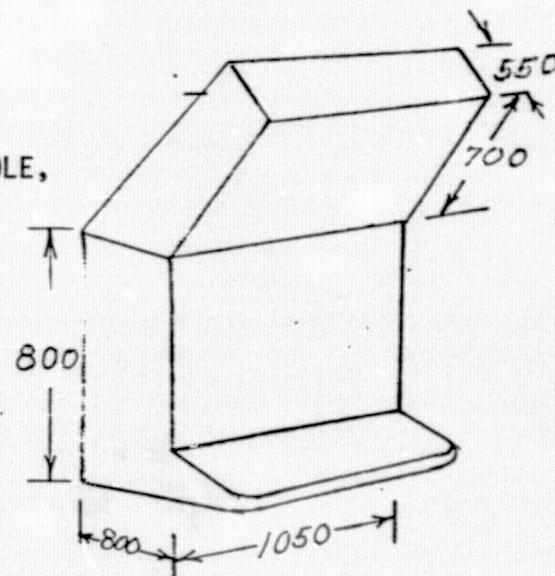


Figure 7-32. Assembly Level Data Sheet — SCB Control Station

1. NAME: EVA AIRLOCK

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES
- o LAUNCH EXPENDABLES

*ASSUMES PUMPDOWN TO 0.5 PSIA

3. SUPPORT REQUIREMENTS

- o HEAT REJECTION (BTU/HR)
- o DATA STORAGE (WORDS)
- o AREA ILLUMINATION (LUM/M²)

NO. REQUIRED: 1

WT. (KG)

454

TBD

TBD

55*

55

VOL. (M³)

4.25

TBD

NONE

54

4. POWER PROFILE

TBD

5. EXTERNAL CHARACTERISTICS 160 CM DIA X 210 CM LONG

LOCATION: INSIDE CSM OR ATTACHED TO OUTSIDE OF CSM

ACCESS: TWO PRESSURE SEALING HATCHES

Figure 7-33. Assembly Level Data Sheet - EVA Airlock

1. NAME: TEST CONTROL STATION

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

180

37

2

--

VOL. (M³)

0.6

0.12

0.003

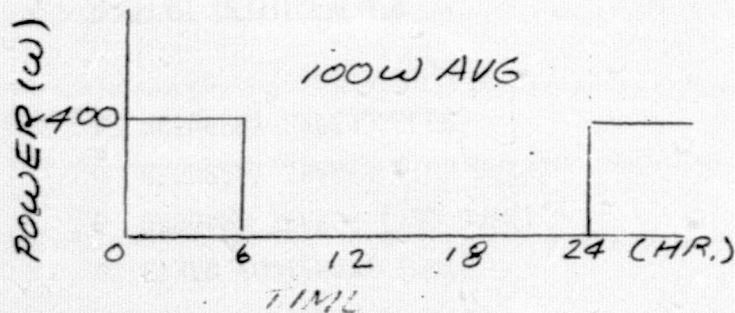
--

3. SUPPORT REQUIREMENTS

- o HEAT REJECTION (BTU/HR)

1×10^3

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: CSM

ACCESS: HINGED/PANELS

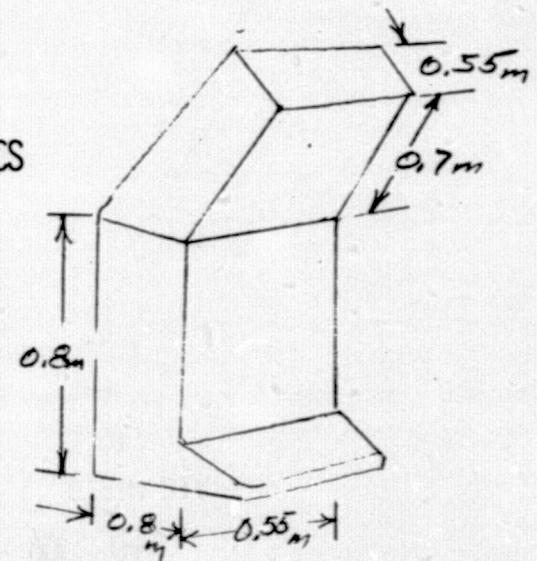


Figure 7-34. Assembly Level Data Sheet — Test Control Station

1. NAME: Ku BAND ANTENNA ASSY

NO. REQUIRED: 2

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

VOL. (M³)

68

0.91M DIA

13

0.01

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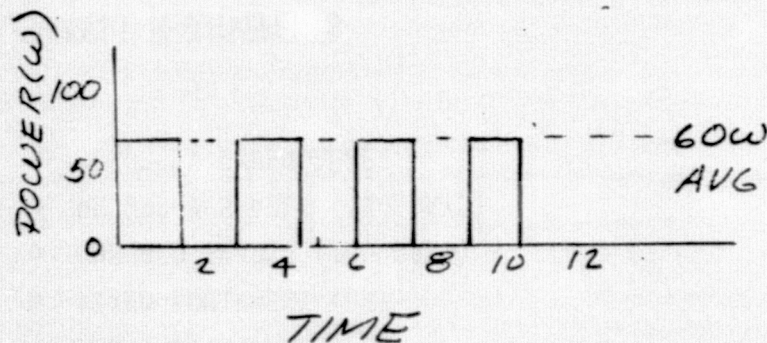
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3. SUPPORT REQUIREMENTS

- o Ku BAND ELECTRONICS ASSY

(XMTTR & RECV.)

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: STRONGBACK

ACCESS: EVA

BOOM
LENGTH
TBD

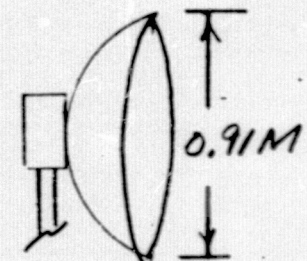


Figure 7-35. Assembly Level Data Sheet - Ku Band Antenna Assembly

1. NAME: TCS (RADIATOR)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)*
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

*INCLUDES FREON FLUID

3. SUPPORT REQUIREMENTS

- o REQUIRES METEROID SHIELD AT
LEAST 0.05 CM THICK

NO. REQUIRED:

WT. (KG)

VOL. (M³)

2.74 KG/M²

N/A

+9 KG FOR END MANIFOLDS

0

0

0

0

4. POWER PROFILE

NONE

5. EXTERNAL CHARACTERISTICS

LOCATION: UNDERSIDE OF
METEROID SHIELD

ACCESS: NONE

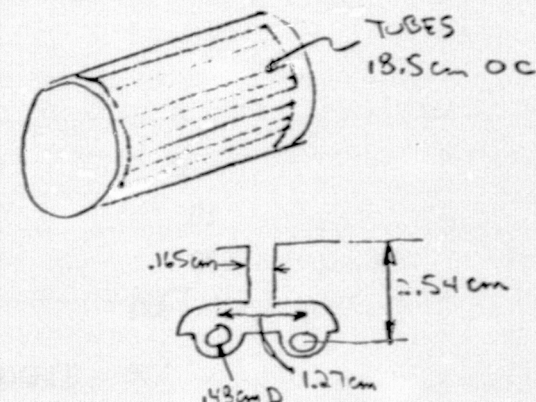


Figure 7-36. Assembly Level Data Sheet - TCS (Radiator)

1. NAME: TCS (COOLANT WATER CIRCULATION)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

NO. REQUIRED: 1/MODULE

WT. (KG)

9.1

4

1

0

VOL. (M³)

.023

.01

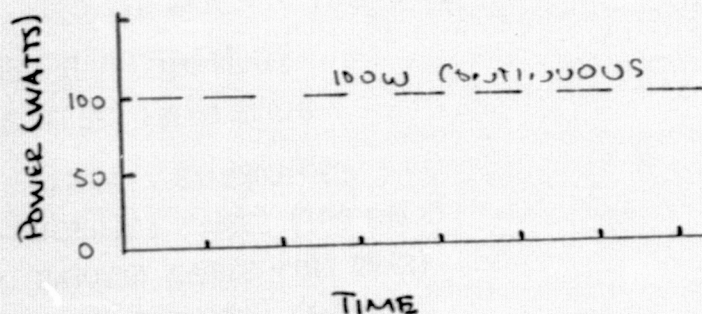
0

0

3. SUPPORT REQUIREMENTS

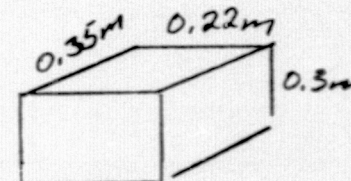
- o CDMS FOR PRESSURE RISE, TOTAL PRESSURE AND ACCUMULATOR LEVEL
- o 1/2" LINES CONNECTING TO WATER LOOP

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: NEAR PENETRATION TO FREON LOOP



ACCESS: REQUIRED ON FOUR SIDES, ALL BUT BACKSIDE AND BOTTOM

Figure 7-37. Assembly Level Data Sheet - TCS (Coolant Water Circulation)

1. NAME: TCS (RADIATOR CONTROL ASSY)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

NO. REQUIRED: 1/MODULE

WT. (KG)

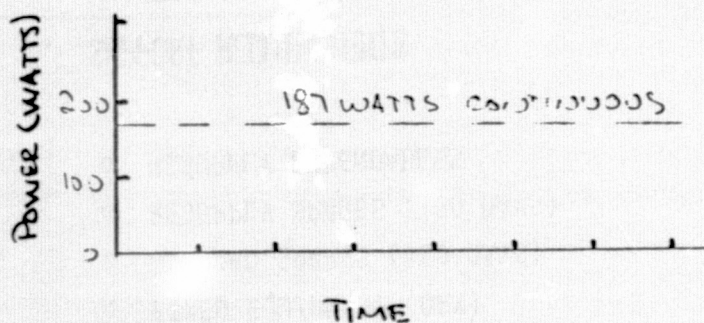
VOL. (M³)

11.4	.05
6.0	.02
1	0
0	0

3. SUPPORT REQUIREMENTS

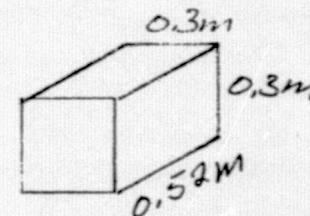
- o CDMS FOR PRESSURE RISE, TOTAL PRESSURE AND ACCUMULATOR LEVEL
- o 3/4" LINES CONNECTING TO FREON LOOP

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: PRESSURE-TIGHT
COMPARTMENT HEAR FREON LOOP
PENETRATION OF PRESSURE SHELL



ACCESS: ALL SIDES BUT BACK
AND BOTTOM

Figure 7-38. Assembly Level Data Sheet — TCS (Radiator Control Assembly)

1. NAME: TCS (RADIATOR CONTROL ASSY)

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

NO. REQUIRED: 1/MODULE

WT. (KG)

VOL. (M³)

19.5	.02
17.0	.01
1.5	0
0	0

3. SUPPORT REQUIREMENTS

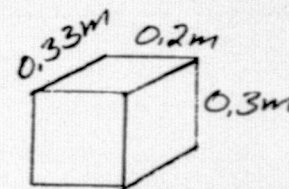
- o CDMS FOR TEMPERATURES AND VALVE POSITION
- o 1/2" LINES FOR WATER LOOP INTERFACE
- o 3/4" LINES FOR FREON LOOP INTERFACE

4. POWER PROFILE

NEGLECTIBLE

5. EXTERNAL CHARACTERISTICS

LOCATION: IN PRESSURE-TIGHT
COMPARTMENT NEAR PRESSURE SHELL
PENETRATIONS FOR FREON LOOP



ACCESS: ALL SIDES EXCEPT
BACK AND BOTTOM

Figure 7-39. Assembly Level Data Sheet — TCS (Radiator Control Assembly)

1. NAME: SATELLITE (MULTI-MISSION MODULE)

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

VOL. (M³)

1070

6

200

0.60

50

0.15

320

0.26

3. SUPPORT REQUIREMENTS

- o POSITIONING PLATFORM
- o LAUNCH CONTROL PANEL
- o DATA ACQUISITION AND CHECKOUT
- o COMMAND AND TRACKING CONSOLE

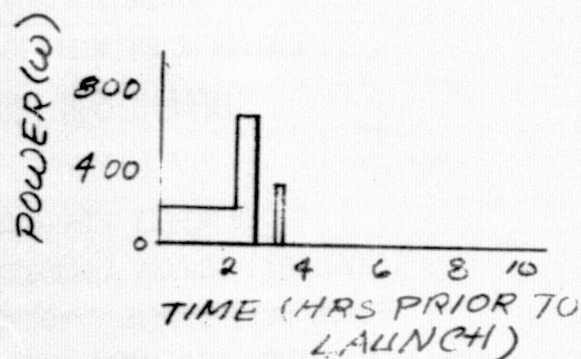
2M³; 50 KG

0.08M³; 1 KG

200 MEASUREMENT

TO 258 KM, 0.35M³; 45 KG

4. POWER PROFILE



5. EXTERNAL CHARACTERISTICS

LOCATION: EXTERNAL TO SCM OR CS

ACCESS: MODULE EXCHANGE MECHANISM AND EVA

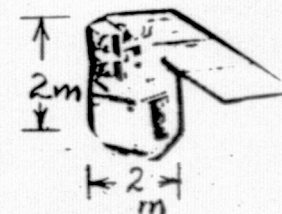


Figure 7-40. Assembly Level Data Sheet — Satellite (Multimission Module)

1. NAME: SATELLITE (BEAM MAPPING)

NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

2363

200

50

320

VOL. (M³)

N/A

0.60

0.15

0.26

3. SUPPORT REQUIREMENTS

- o POSITIONING PLATFORM
- o LAUNCH CONTROL PANEL
- o DATA ACQUISITION AND CHECKOUT
- o COMMAND AND TRACKING CONSOLE

4. POWER PROFILE

SEE MULTI MISSION
MODULE SATELLITE

5. EXTERNAL CHARACTERISTICS

LOCATION: EXTERNAL

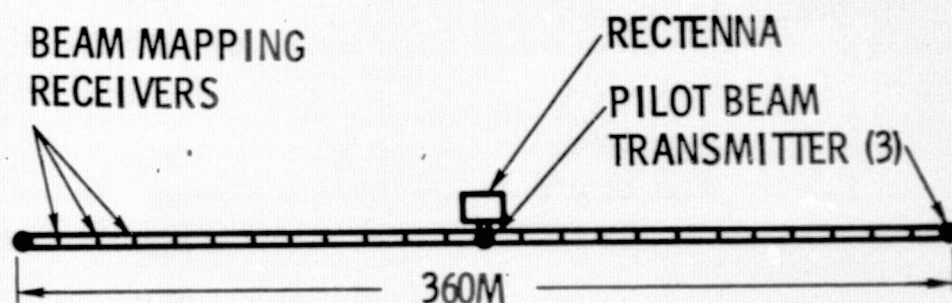


Figure 7-41. Assembly Level Data Sheet — Satellite (Beam Mapping)

1. NAME: SATELLITE (COMMAND AND TRACKING CONSOLE) NO. REQUIRED: 1

2. PHYSICAL CHARACTERISTICS

- o FIXED EQUIPMENT (DRY)
- o ONBOARD SPARES (180 DAYS)
- o RESUPPLY SPARES (180 DAYS)
- o RESUPPLY EXPENDABLES

WT. (KG)

VOL. (M³)

45

0.12

9

0.03

--

--

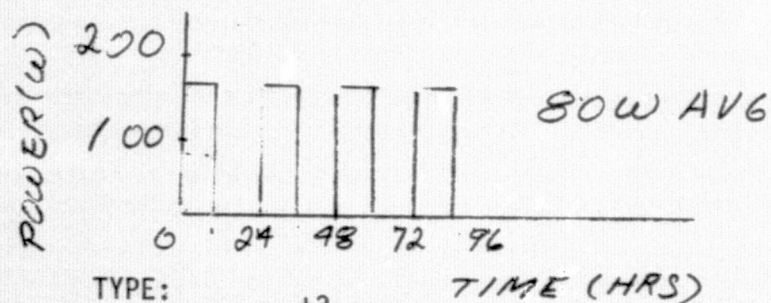
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3. SUPPORT REQUIREMENTS

- o Ku BAND RADAR
- o DATA MANAGEMENT SYSTEM
- o GUIDANCE SYSTEM
- o LASER TRACKING SYSTEM

4. POWER PROFILE



TYPE:
REG. DC: 28 ⁺³₋₂
UNREG. DC: --
OTHER: 15, 5 VDC

5. EXTERNAL CHARACTERISTICS

LOCATION: INTERNAL

ACCESS: HINGED PANELS

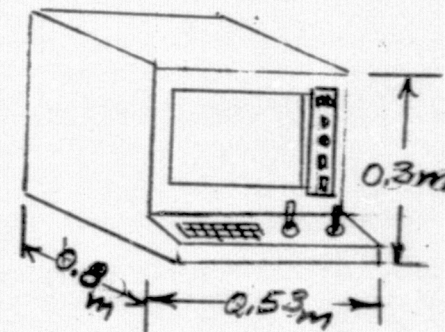


Figure 7-42. Assembly Level Data Sheet — Satellite (Command and Tracking Console)

Section 8 MISSION OPERATIONS

8.1 CONSTRUCTION OPERATIONS FLOWS

The procedure followed in analyzing construction was to first take the preliminary design layout for each item of mission hardware and packaging (for delivery) layouts and visualize how the part would be constructed. This was done in conjunction with the designer and the layouts modified where problems were identified. A detailed flow logic was then developed with each step conceived to be logical (and feasible) sequel to its immediate predecessor. As these logic flows were assembled, design modifications deemed desirable to simplify the construction process were proposed and, as before, coordinated with the designer. Upon achieving an acceptable flow, each event was analyzed to determine how long it would take, how much extravehicular activity (EVA) translation distance would be involved, the required crane reach, etc. These data were then entered on the flows as appropriate.

The time required to construct mission hardware tends to be critical in making decisions with respect to the best construction methods, hardware configuration, etc. In establishing the time to perform various tasks, specific analyses were used (e. g., use of crane dynamics data to compute transfer times) where possible; in others, Skylab experience was used. In some cases, direct estimates were made based on ground experience extrapolated to zero-g environment. The most time-consuming tasks were then identified, and sensitivity analyses were performed to determine the criticality of our analyses or estimates. For those tasks deemed critical, additional analyses were performed and the time estimates on the flows updated as appropriate.

The resultant flows developed in Part 3 are included in this appendix and consist of the following:

8.1 Power Platform 456 kW (Process times for 250 kW also noted)
(Figure 8-1)

- Ladder concept utilizing on orbit fabrication constructed on a fixed work station assembly fixture
- The same as above but constructed on the traveling work station construction platform
- Power platform assembly only concept using a retractable pallet
- Same as the above using construction platform

NOTE: See 4.1 and 4.2 for times associated with power platform deployment and power platform fabrication with auto assy.

8.2 100m Radiometer (Figure 8-2)

- Assembly using fixed work station concept employing a telescoping beam
- Assembly on the construction platform

8.3 TA-1 (Figure 8-3)

- Deployment using fixed work station concept employing an extendable pallet
- Fabrication with automatic assembly detailed event sequence (from Part 2)

8.4 TA-2 (Figure 8-4)

- TA-2 Deployment and assembly using fixed work station employing a fixed standoff TA-2 antenna only deployment
- TA-2 Fabrication with automatic assembly detailed event sequence (from Part 2)

8.5 Beam Mapping Satellite (Figure 8-5)

- Beam Mapping Satellite fabrication using fixed work station concept.

(1.1)

456 KW POWER PLATFORM FAB & ASSY LADDER CONCEPT USING ASSY FIXTURE

HOURS (MIN)

[TIME FOR 0.1 M/MIN]
FAB RATE

<CHERRY PICKER TRANSLATION DISTANCE>

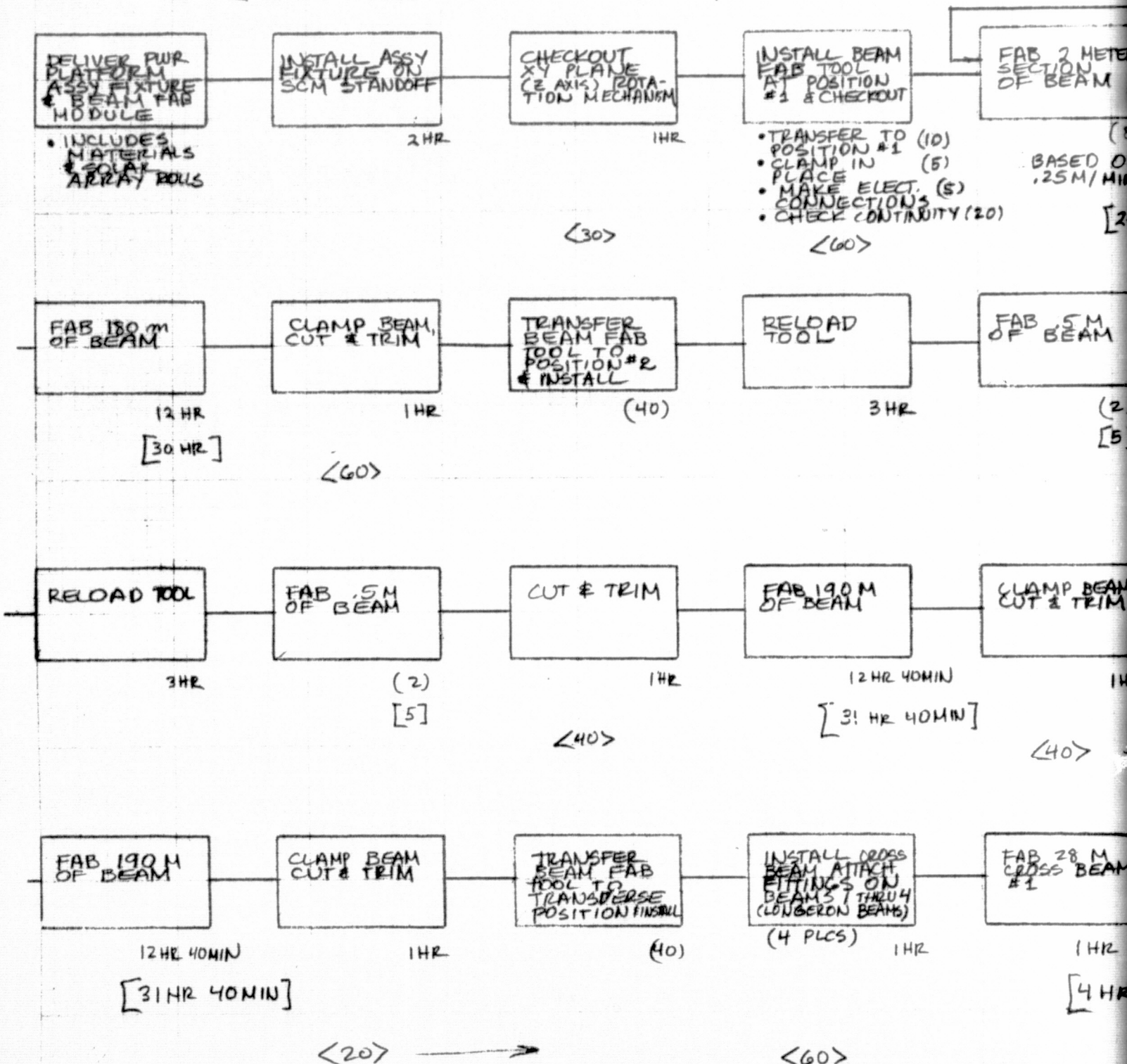
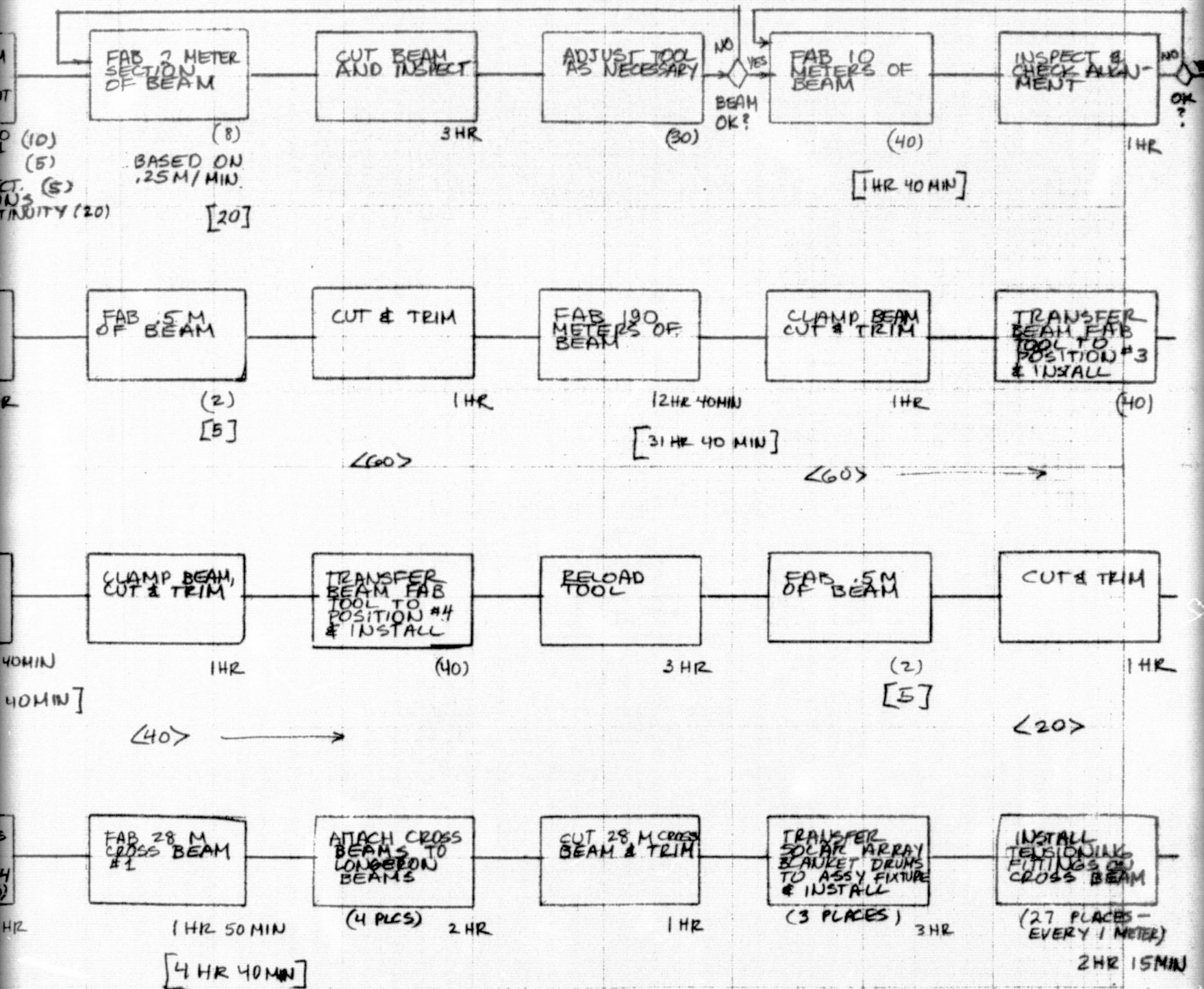
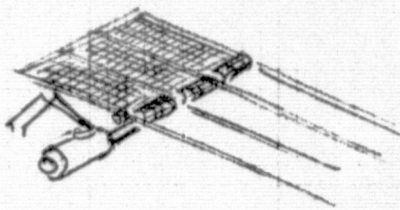


Figure 8-1. Power Platform (Sheet 1 of 7)

K&E

10 X 10 TO THE INCH

FOLDOUT FRAME



FOLDOUT FRAME 2

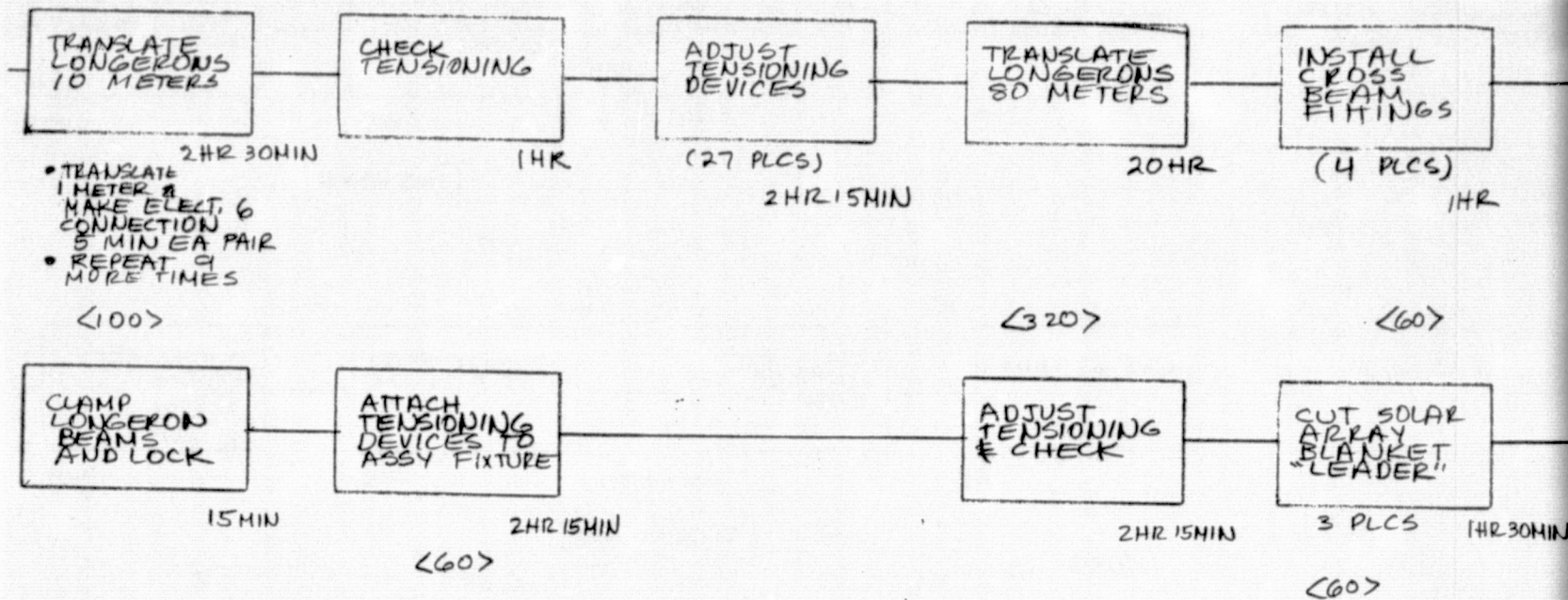
PAGE 1 OF 2

REVISED 5-7-77

REH 5-4-77

KEUFFEL & ESSER CO.
MADE IN U.S.A.

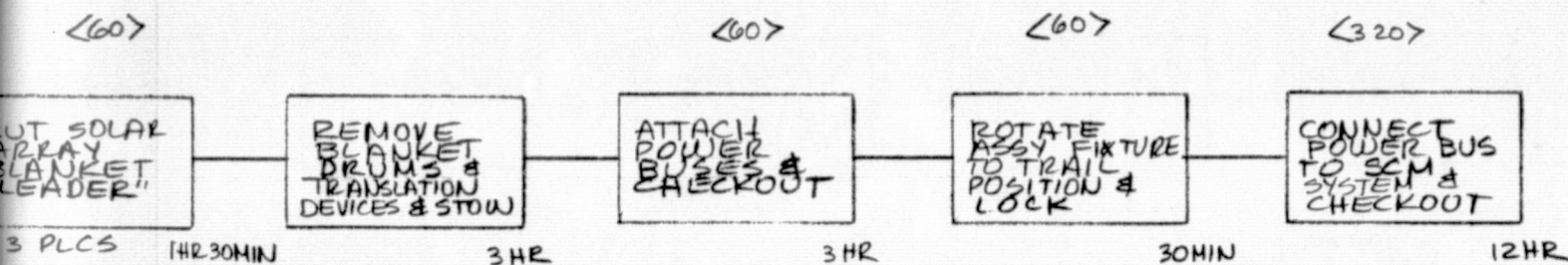
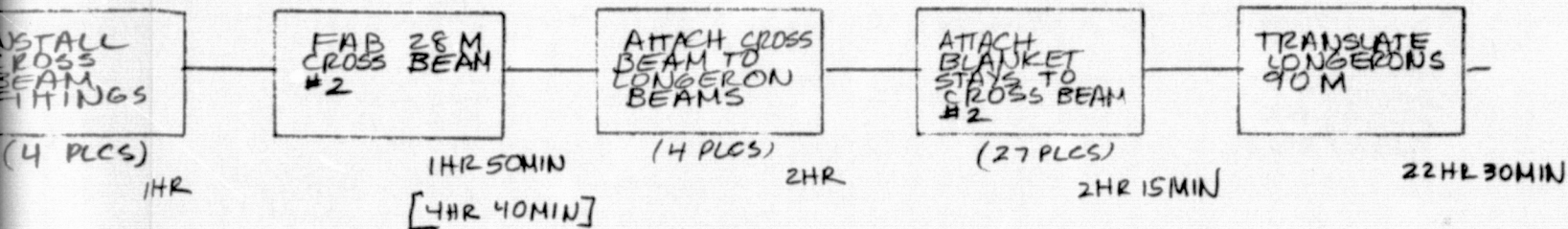
456 KW POWER PLATFORM FAB & ASSY LADDER CONCEPT



$$167 \text{ HRS} = 28 \text{ SHIFTS} = 2\frac{1}{3} \text{ WEEKS}$$

$$[249 \text{ HRS} = 42 \text{ SHIFTS} = 3\frac{1}{2} \text{ WEEKS}]$$

$$\text{for 250 KW array} = 194 \text{ hrs} = 32.33 \text{ shifts} = 2.7 \text{ weeks}$$



weeks

(1.2)

456 KW POWER PLATFORM (PP) FAB & ASSY USING CONSTRUCTION PLATFORM

HOURS (MIN)

BASED ON 0.11 M/SEC

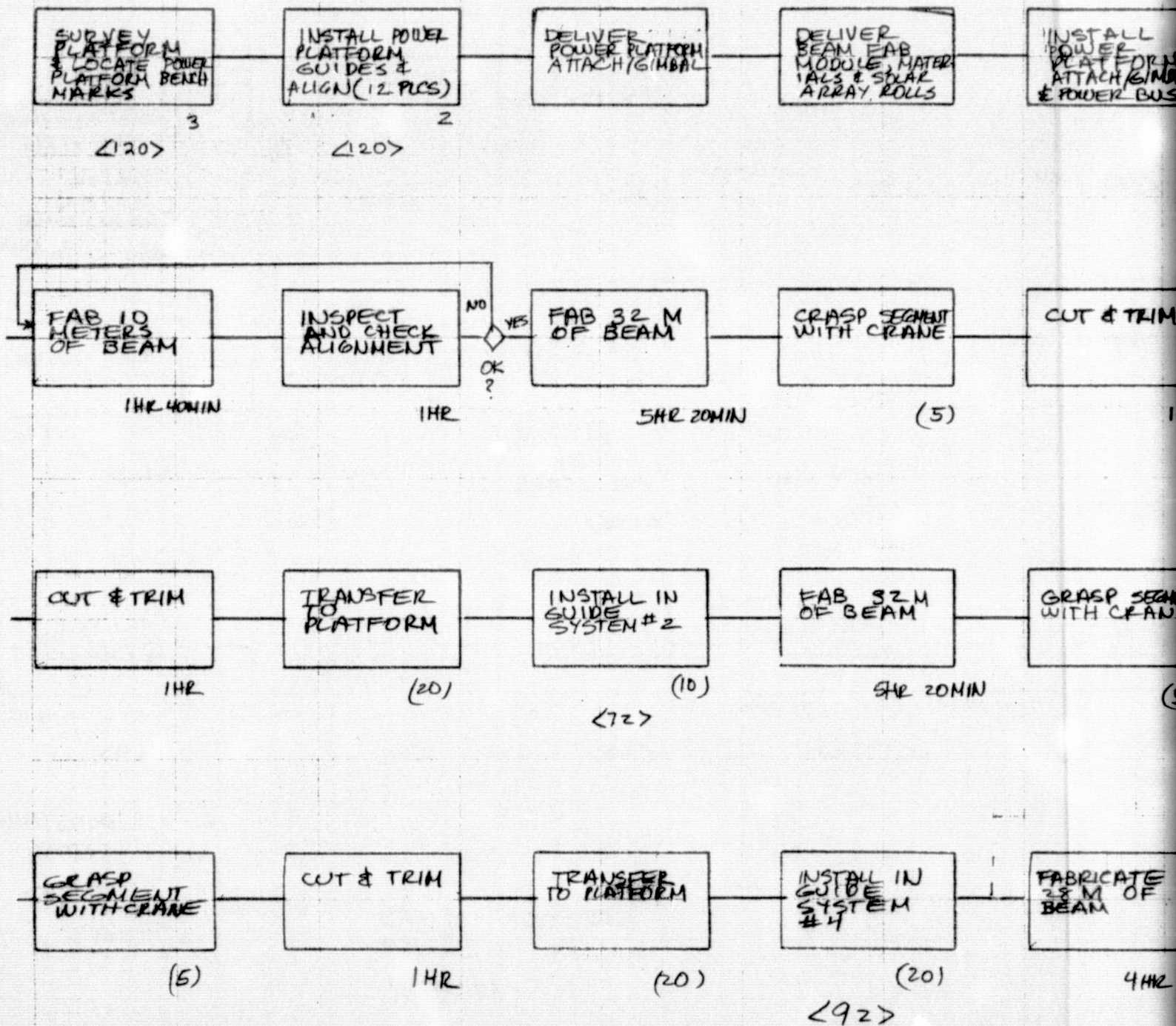
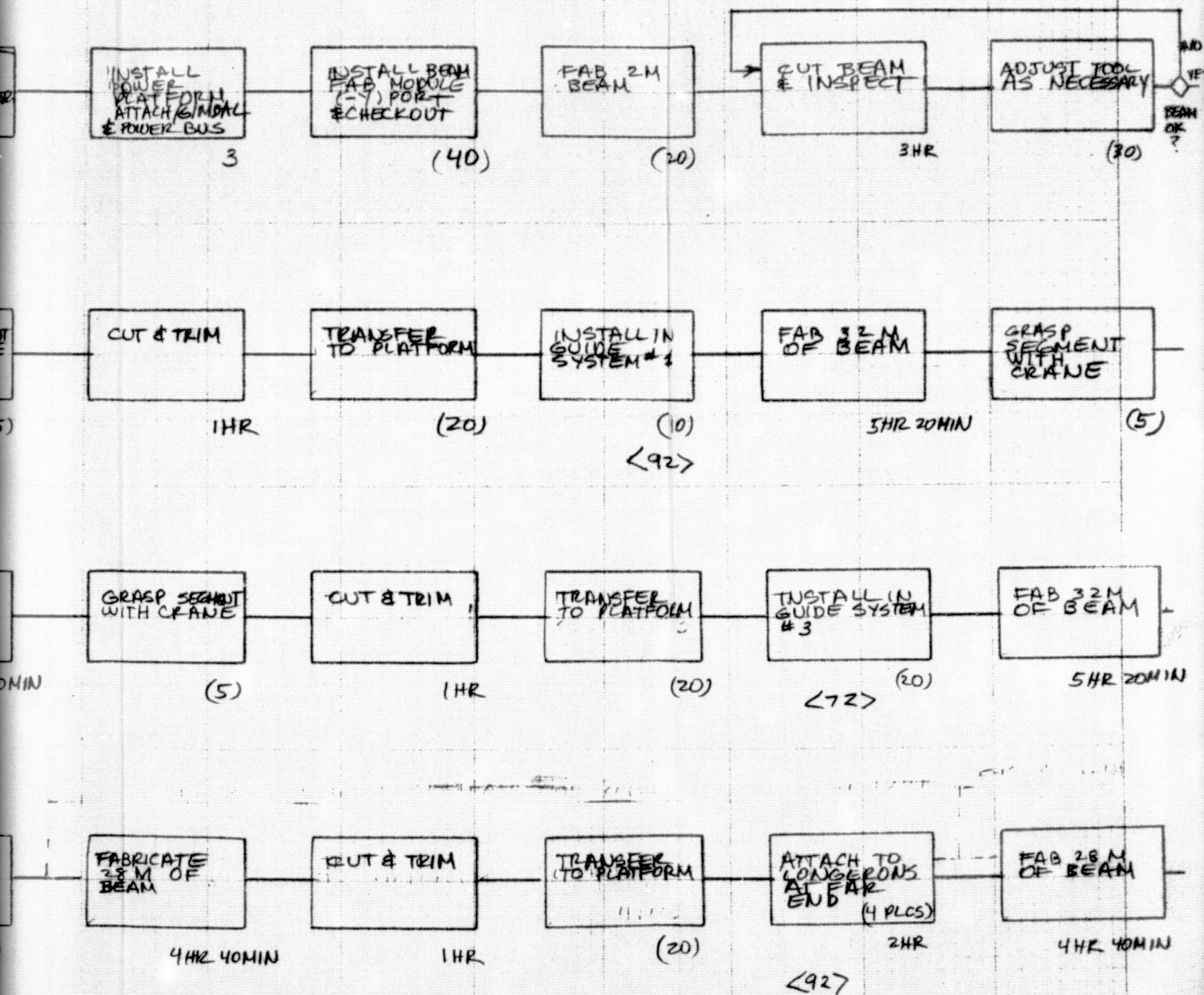


Figure 8-1. Power Platform (Sheet 3 of 7)

K&E

ALBANY, N.Y. 12207-1000
TO 5645
10 X 10 TO THE INCH

FOLDOUT FRAME



FOLDOUT FRAME 2

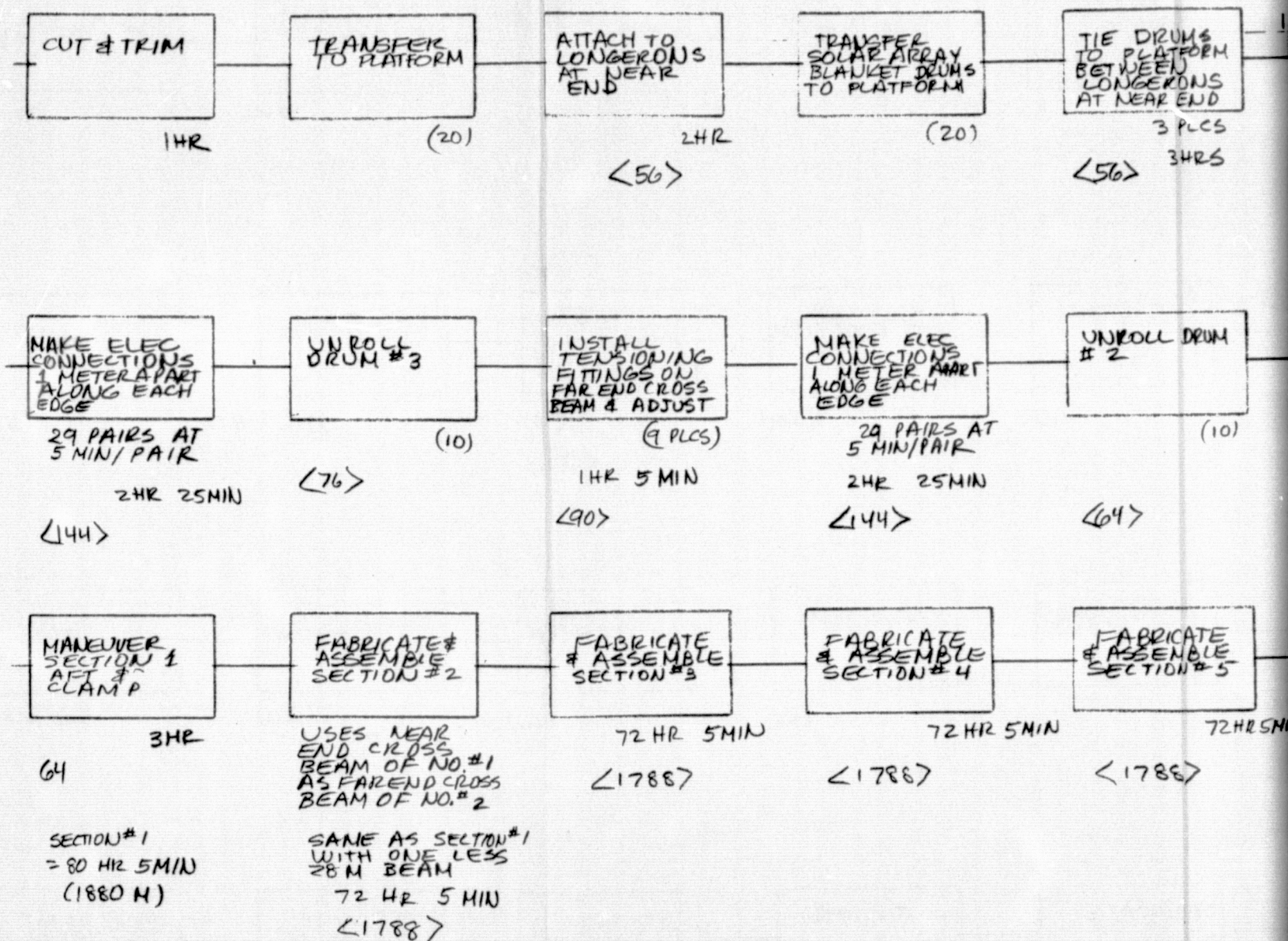
REVISED 5-18-77

1 OF 2

REH 5-6-77

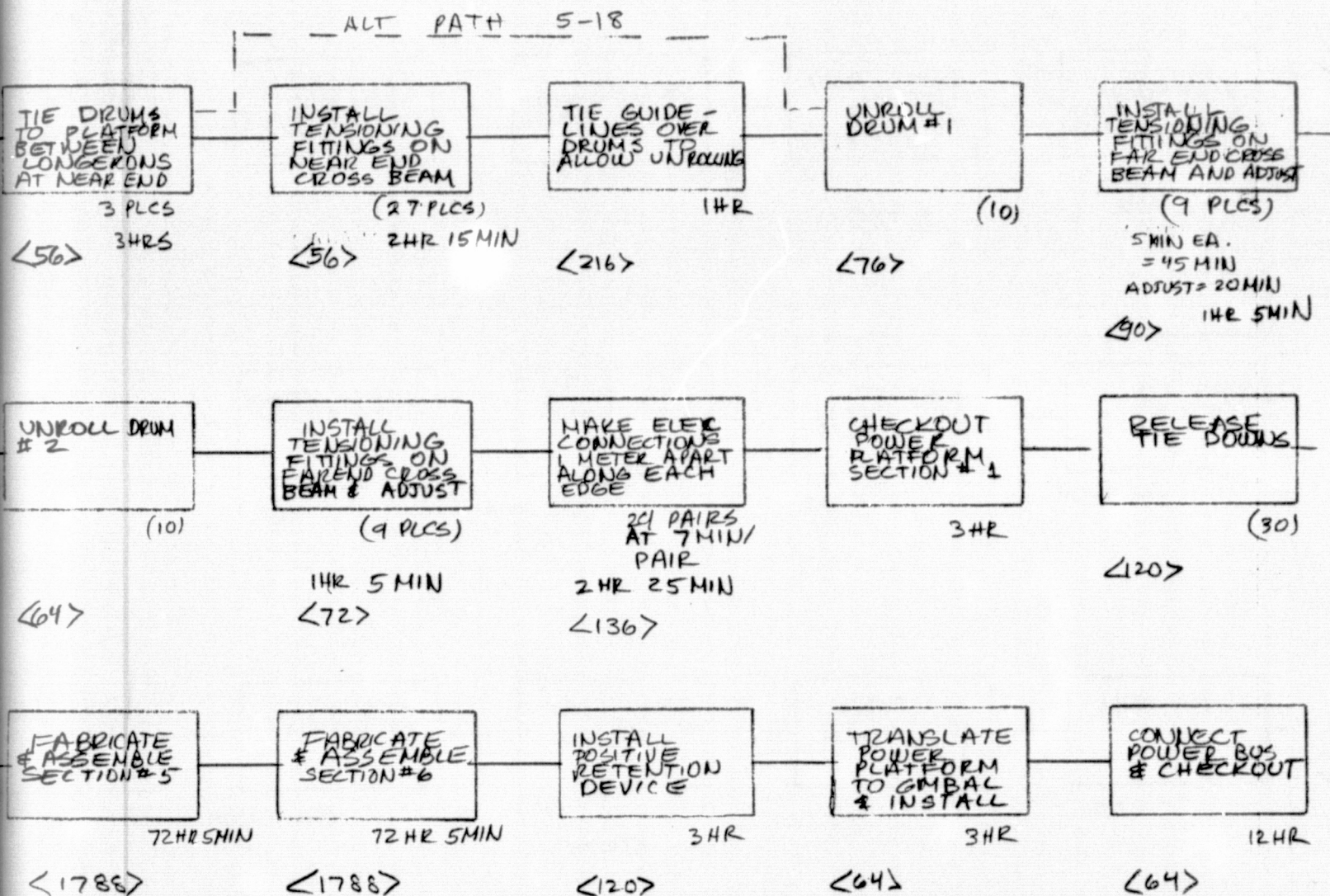
KELUFFEL PRESSER CO.
MADE IN U.S.A.

456 KW POWER PLATFORM
FAB & ASSY USING CONSTRUCTION
PLATFORM



FOLDOUT FRAME 3

Figure 8-1. Power Platform (Sheet 4 of 7)



Σ = 458 HOURS 30MIN
= 76 SHIFTS
= 6 ¹/₃ WEEKS

<11 KM>

REVISION = 419 HRS
5-18 = 5 ³/₄ WEEKS
(8.5 KM)

REVISED 5-18

20F2

REH 5-6-77

FOLDOUT FRAME 4

1.3 456 KW POWER PLATFORM
ASSEMBLY USING RETRACTABLE
PALLET

(REVISED)
5-24-77

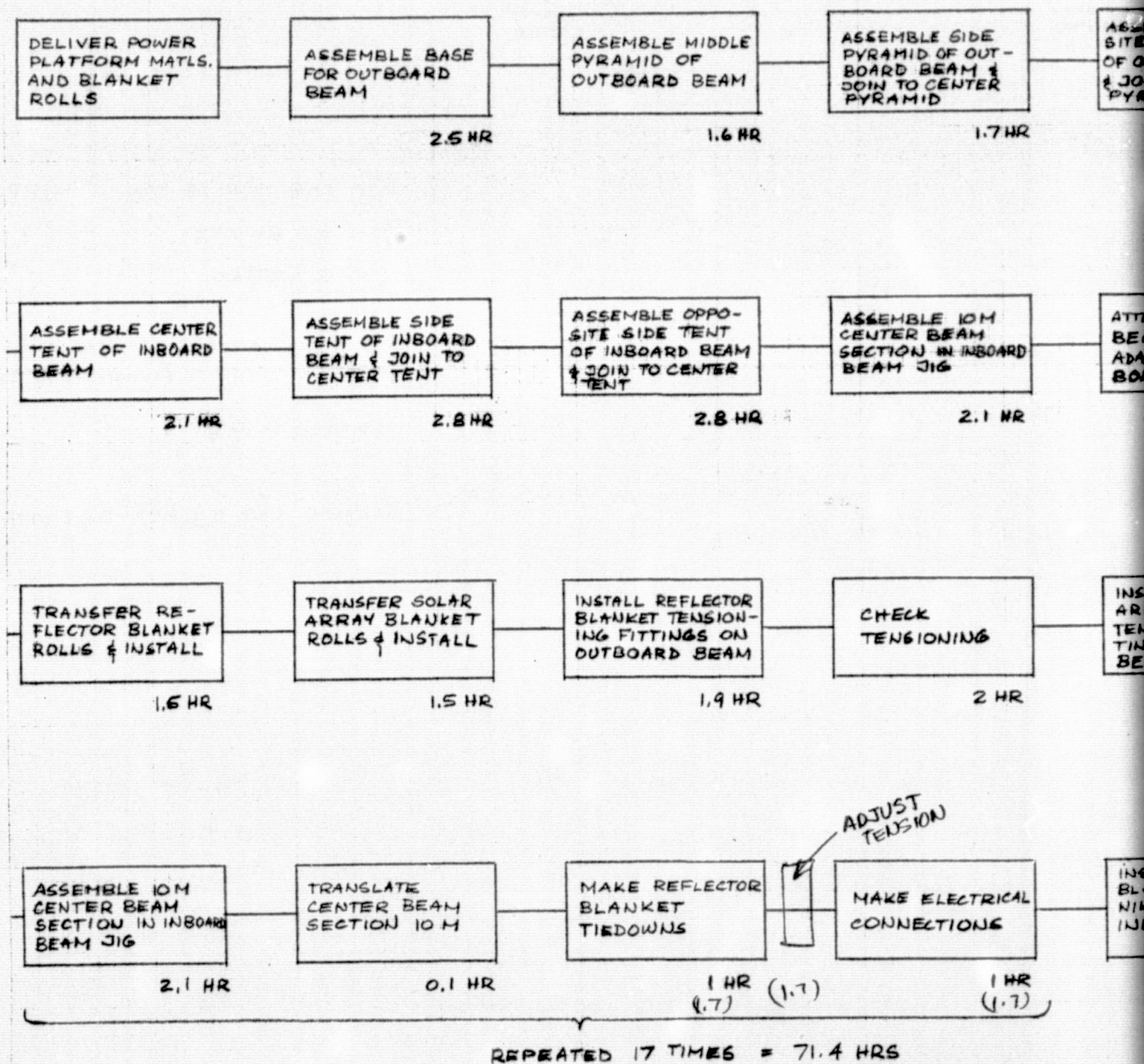
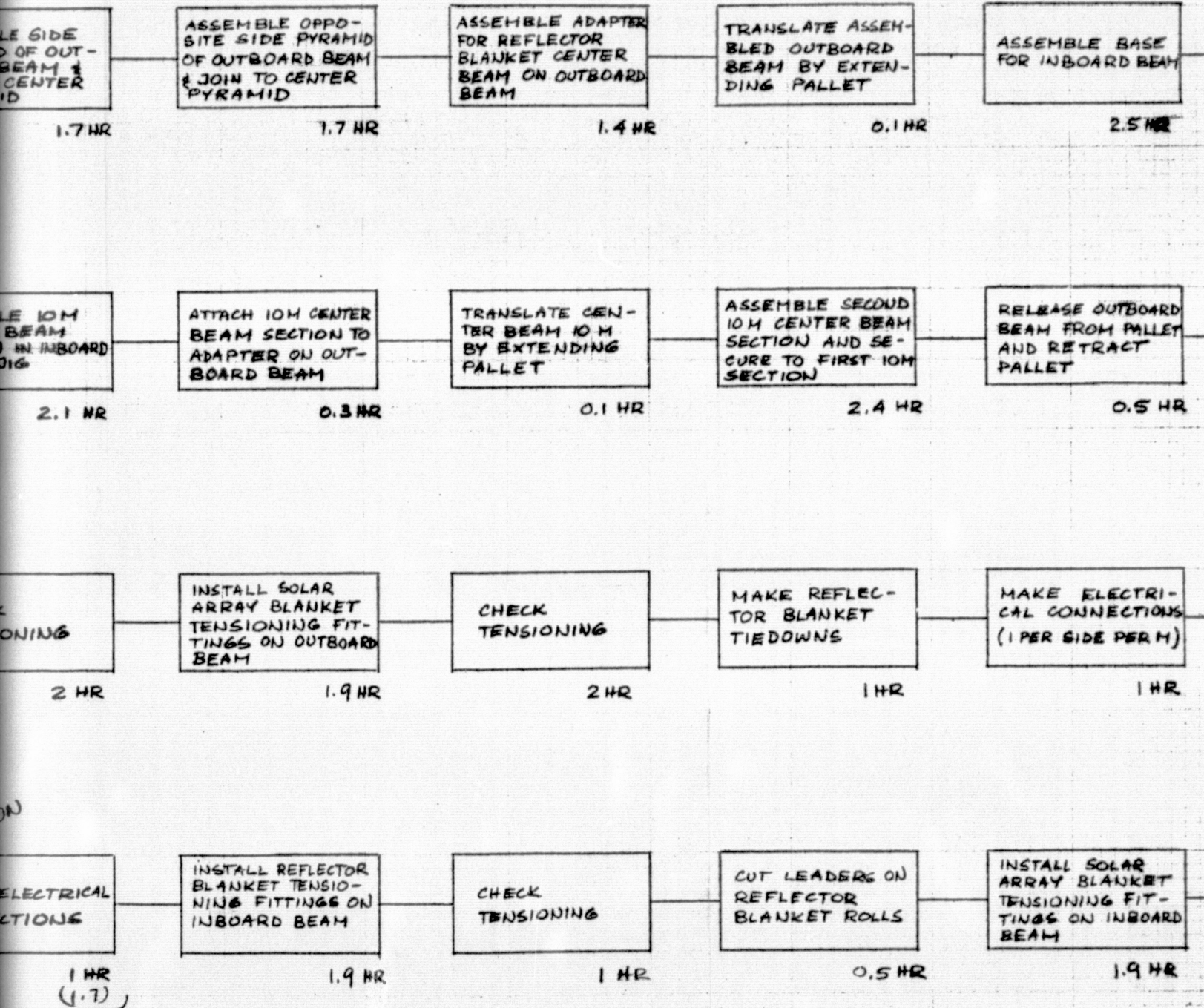
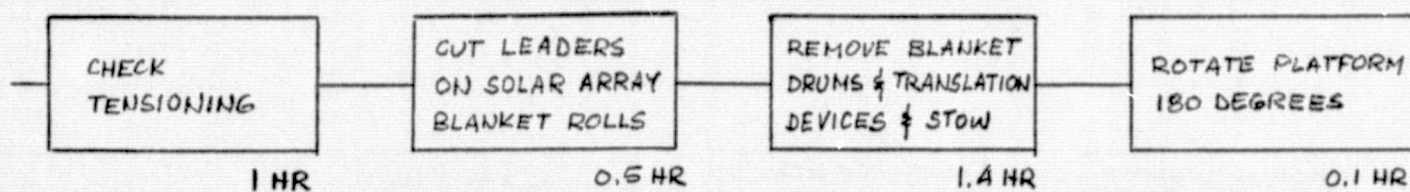


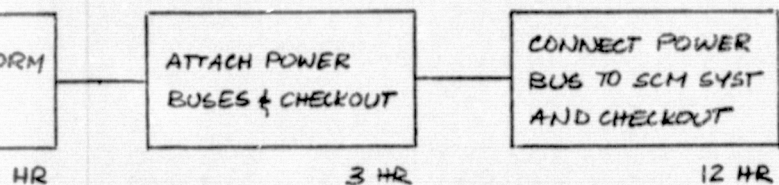
Figure 8-1. Power Platform (Sheet 5 of 7)

FOLDOUT FRAME





FOLDOUT FRAME 3

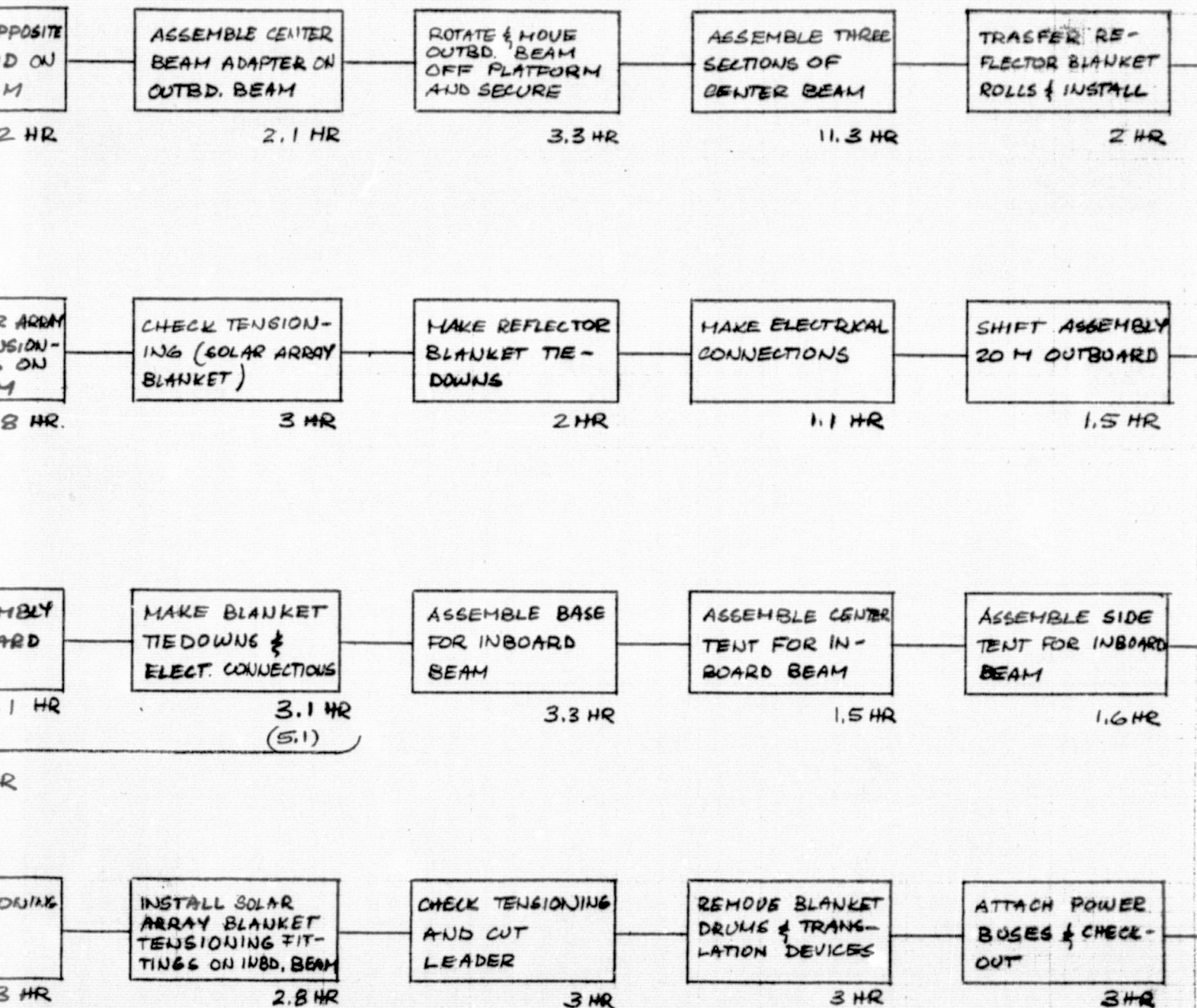


$\Sigma = 136 \text{ HRS } 18 \text{ MIN.} \quad (188.7)$
 $= 23 \text{ SHIFTS} \quad (32)$
 $\approx 2 \text{ WEEKS}$

EVA TRANSLATION DISTANCE = 5.9 KM.
(RIDING IN CHERRY PICKER PLATFORM)

TASKS - 1981

250 KW = 130.3 HRS = 21.7 SHIFTS



Σ = 188 HRS. 54 MIN
 = 32 SHIFTS (38)
 = 2.7 WEEKS
 EVA 11.4 KM

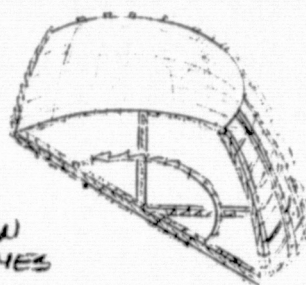
TASKS=1734
 KLUFFEL-BESSER CO.
 MADE U.S.A.

FOLDOUT FRAME 2

(2.1) 100 M RADIOMETER ASSEMBLY
(CHERRY PICKER SUPPORTED)

1 HRS (MIN)

[] INDICATES REVISED TIMES
BASED ON REVISED EVALUATION
OF TRANSLATION & ROTATION TIMES



< EVA DISTA

○ TASKS

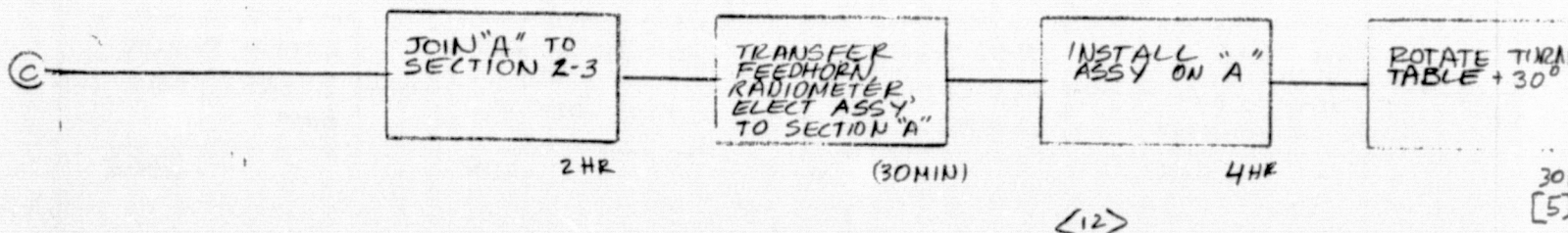
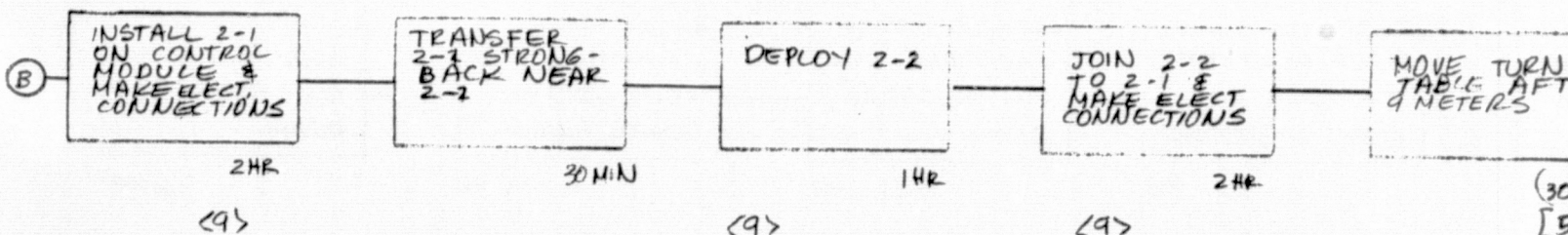
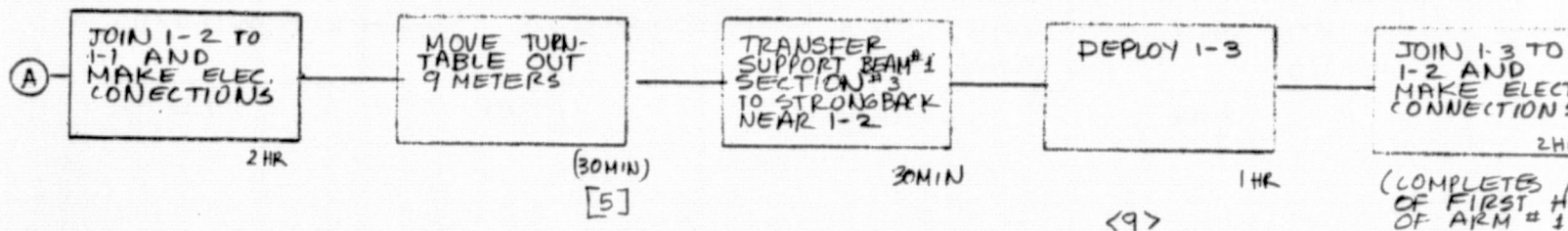
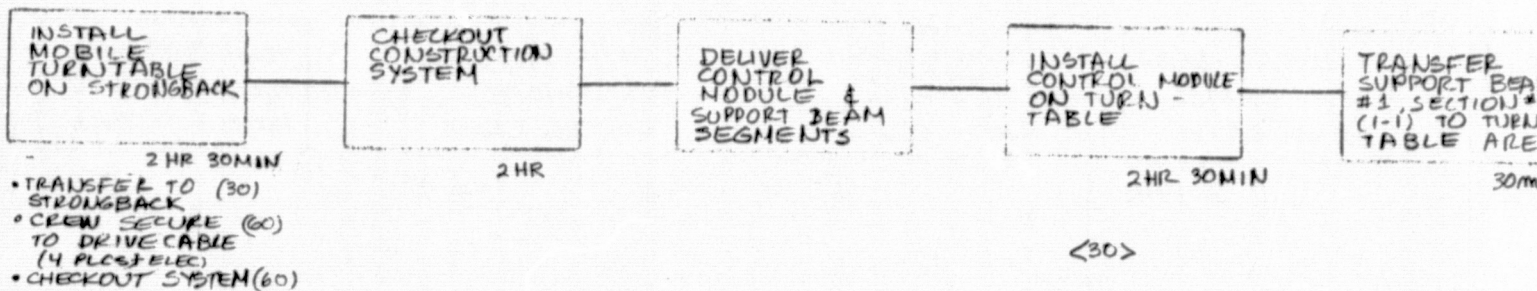
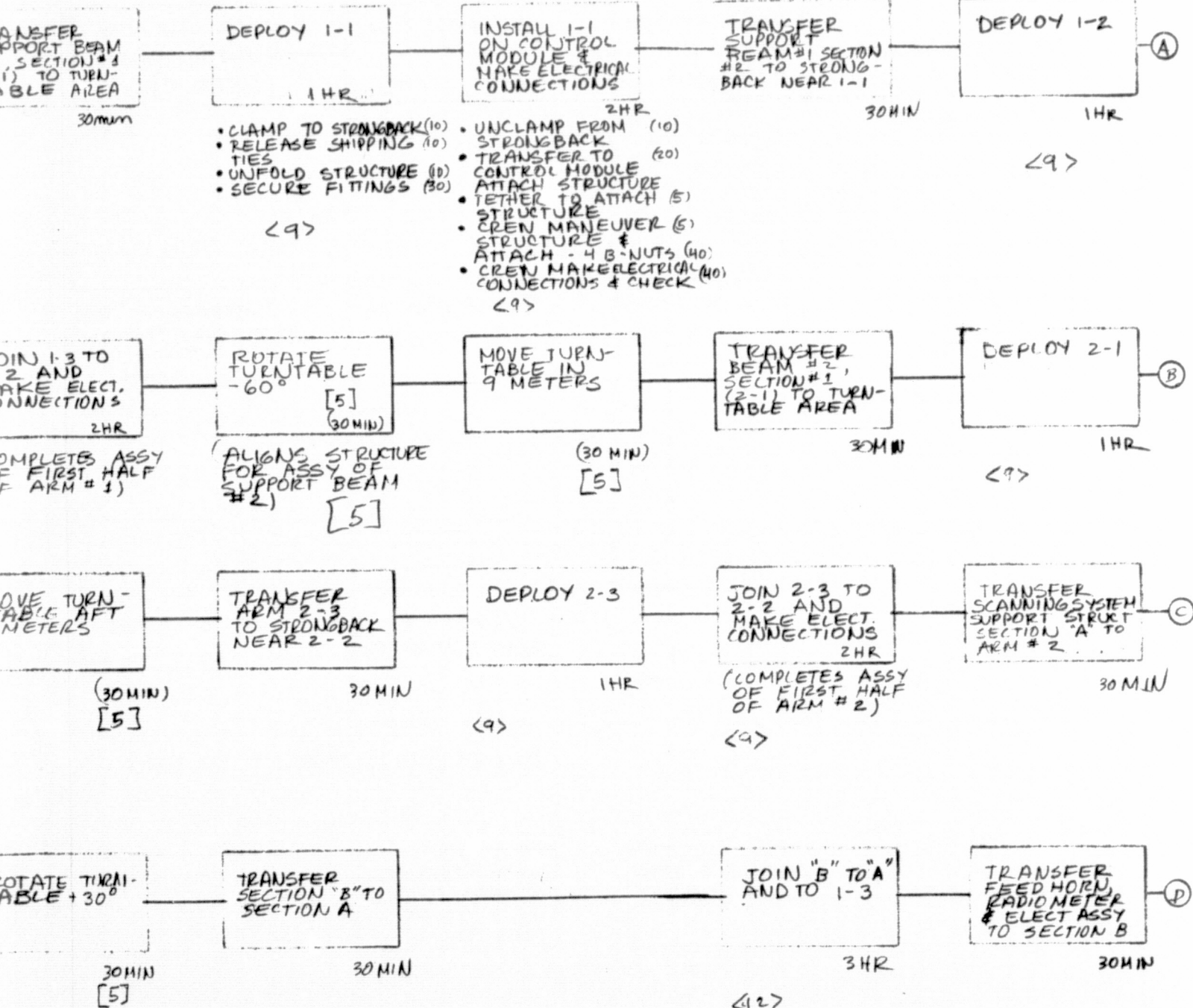
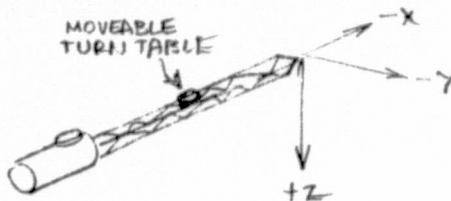


Figure 8-2. 100m Radiometer (Sheet 1 of 5)

VA DISTANCE)

TASKS



FOLDOUT FRAME

1 OF 3

REF 3-21-77

100 M RADIOMETER ASSY (CONT.)

FOLDOUT FRAME

Handwritten mark resembling a stylized 'B' or '8'.

ORIGINAL PAGE IS
OF POOR QUALITY

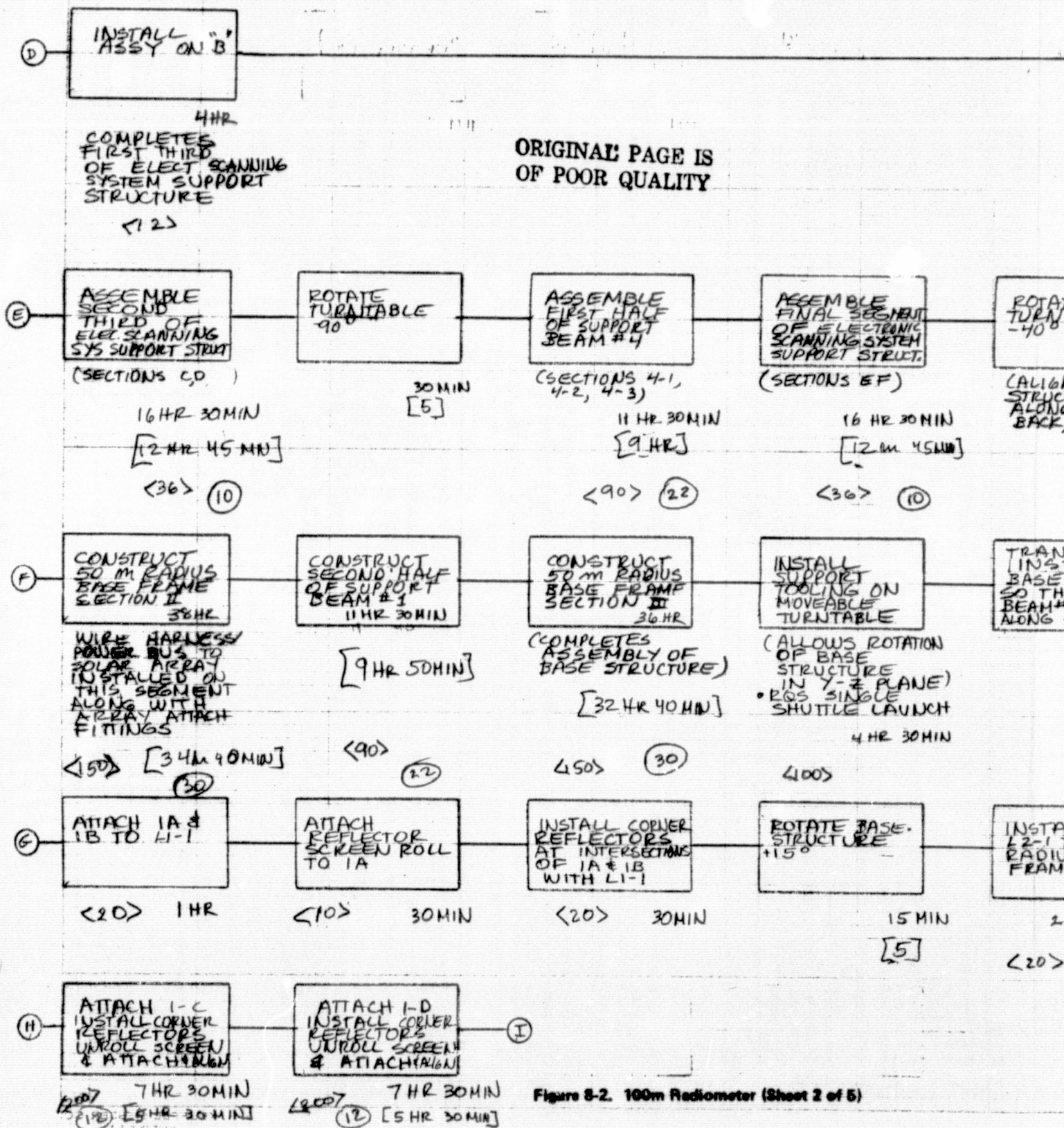
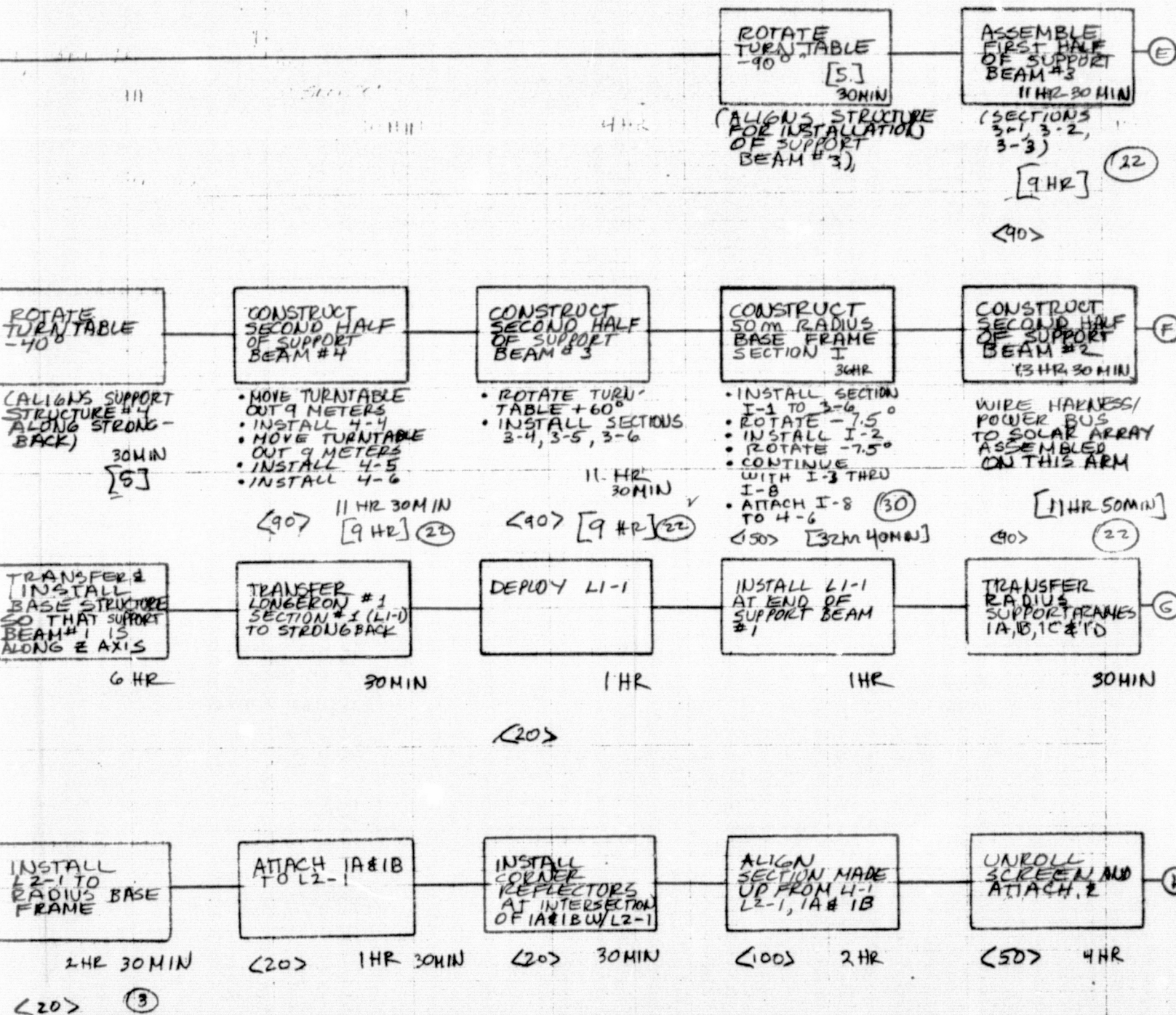


Figure 8-2. 100m Radiometer (Sheet 2 of 5)

FOLDOUT FRAME 2

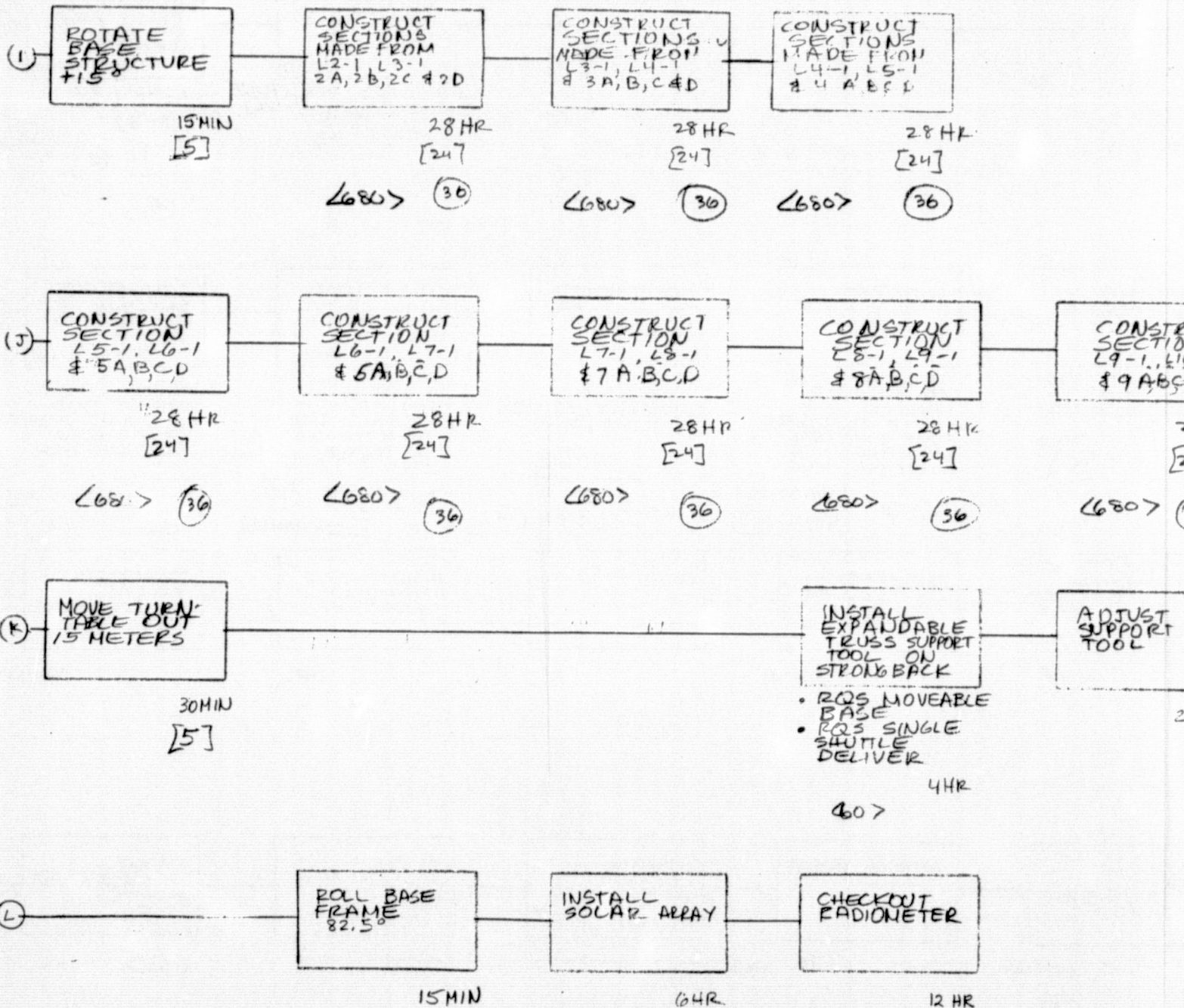


2 OF 3

REH 3-21-77

100 m RADIOMETER ASSY (CONT.)

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OF POOR QUALITY

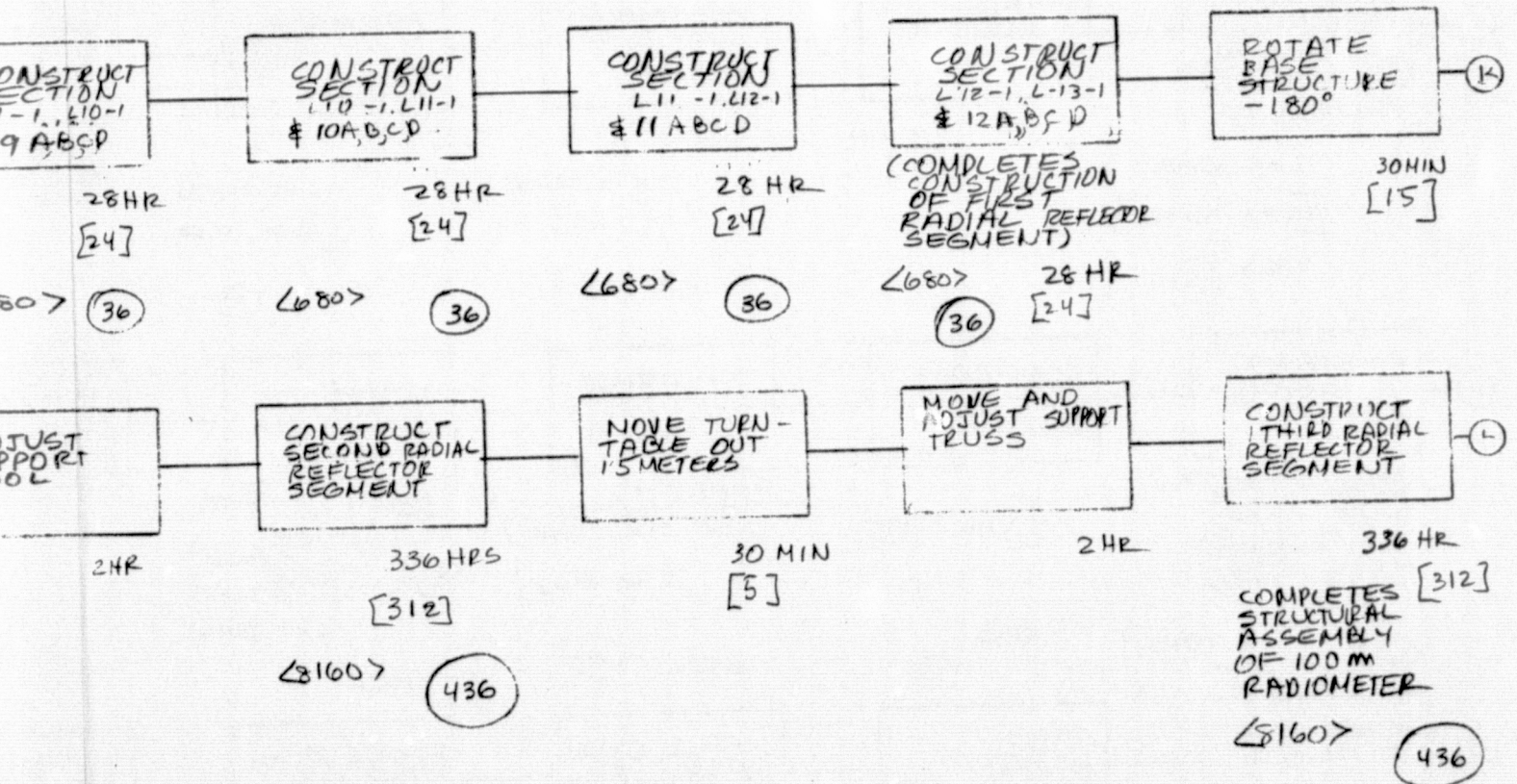


FOLDOUT FRAME 3

1238 HRS
206 SHIP

Figure 8-2. 100m Radiometer (Sheet 3 of 5)

ORIGINAL PAGE 1
OF POOR QUALITY



3 HRS
10 SHIFTS

[1047 HRS]
[175 SHIFTS]

1840

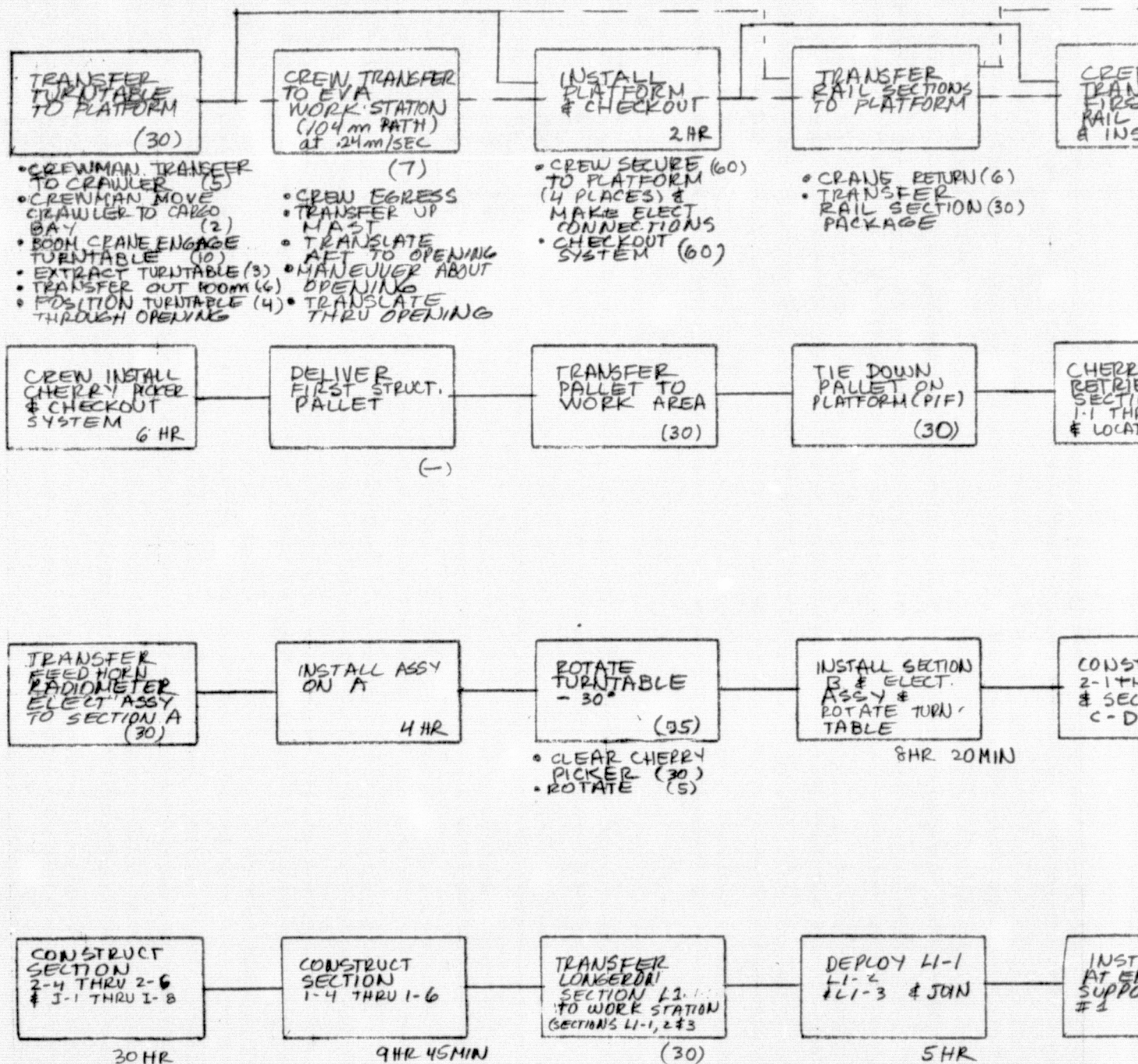
<26 KM>

(2413)

FOLDOUT FRAME 4

3 OF 3
REH 3-22-77

2.2 100 M RADIOMETER ASSY
USING A PLATFORM
CONSTRUCTION

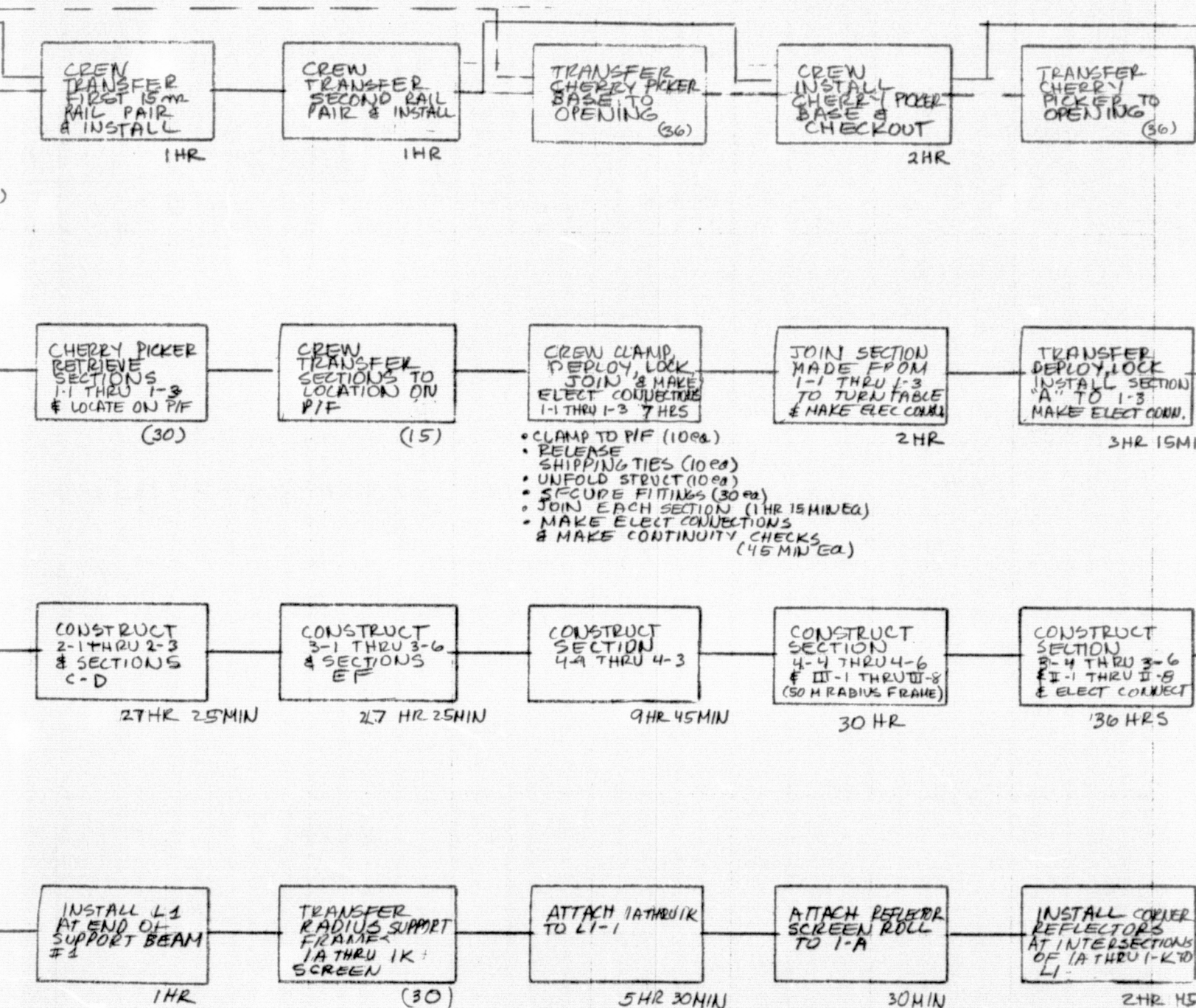


FOLDOUT FRAME

Figure 8-2. 100m Radiometer (Sheet 4 of 5)

ORIGINAL PAGE IS
OF POOR QUALITY

ORIGINAL PAGE IS
OF POOR QUALITY



FOLDOUT FRAME

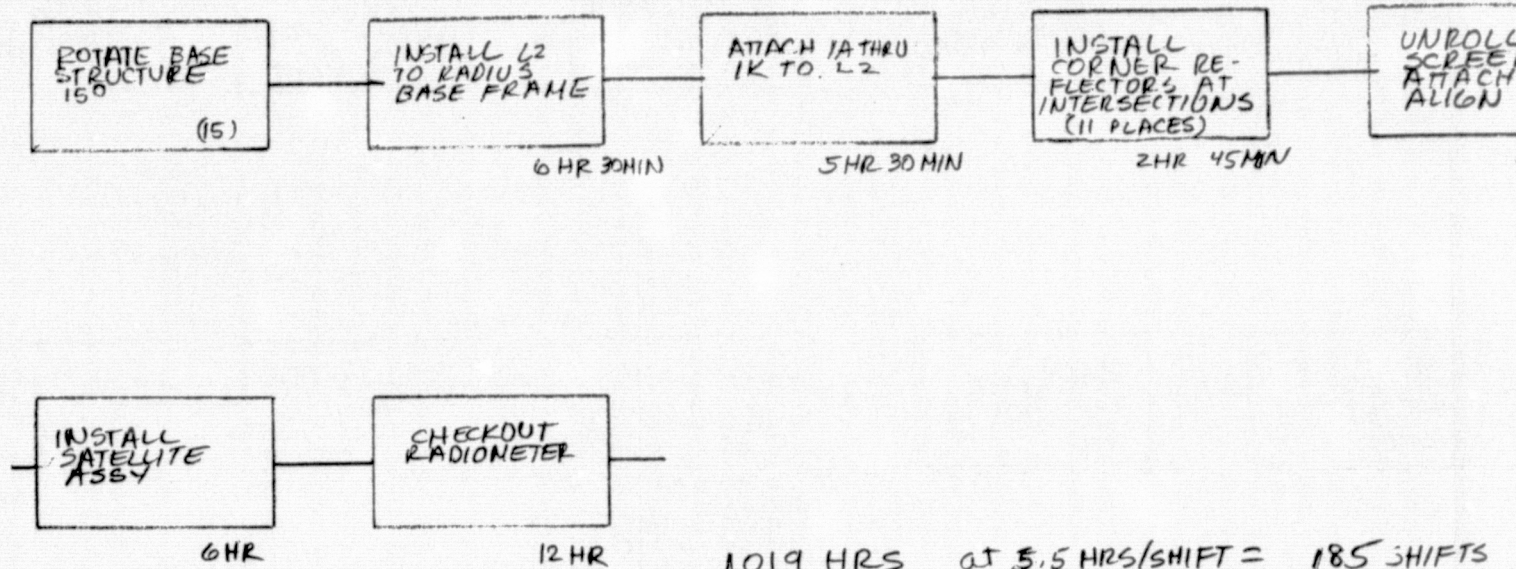
10F2

REH 4-7-77

8-13

100 M RADIOMETER ASSY USING PLATFORM

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OF POOR QUALITY

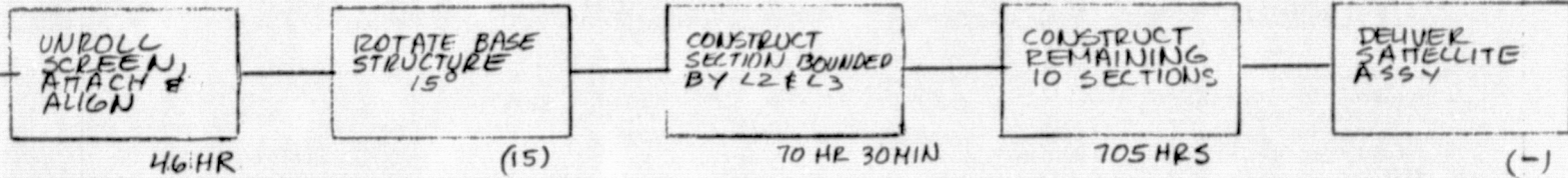


1019 HRS AT 5.5 HRS/SHIFT = 185 SHIFTS

EVA L26 K
TASKS (2

FOLDOUT FRAME 3

ORIGINAL PAGE IS
OF POOR QUALITY



85 SHIFTS

EVA $26 \text{ KM} + 185 \times 50 = 33.5 \text{ KM}$
TASKS (2239)

FOLDOUT FRAME 4

(3.1) TA-1 DEPLOYMENT WITH MINIMUM ASSEMBLY
(CHERRY PICKER UTILIZED)

TRANSLATION TIME

EVA 1.41/SEC
CRANE 0.7 - 2.3 ft/SEC [500' - 5000']
(REVISED TIMES BASED ON PESSA'S ANALYSIS.)

ARRAY SEGMENT ≈ 400

NEXT 10

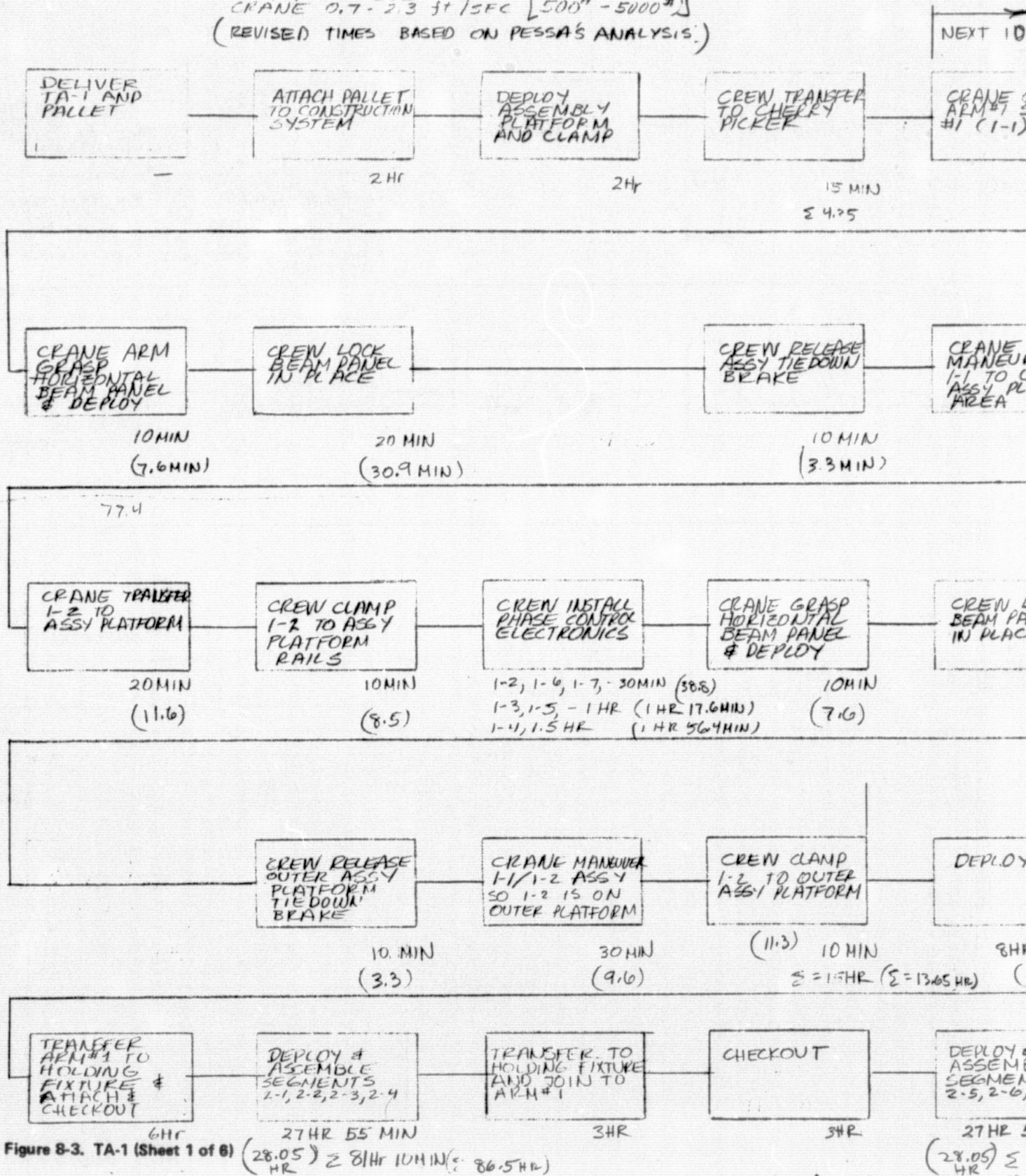


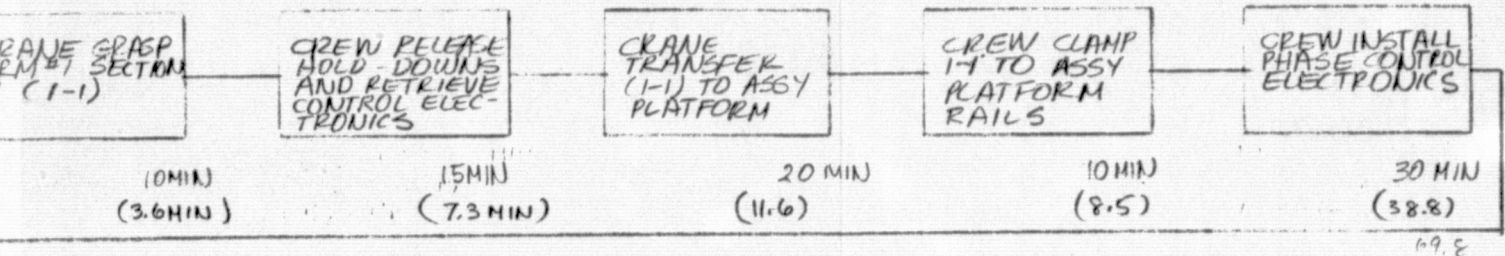
Figure 8-3. TA-1 (Sheet 1 of 6)

ORIGINAL PAGE IS
OF POOR QUALITY

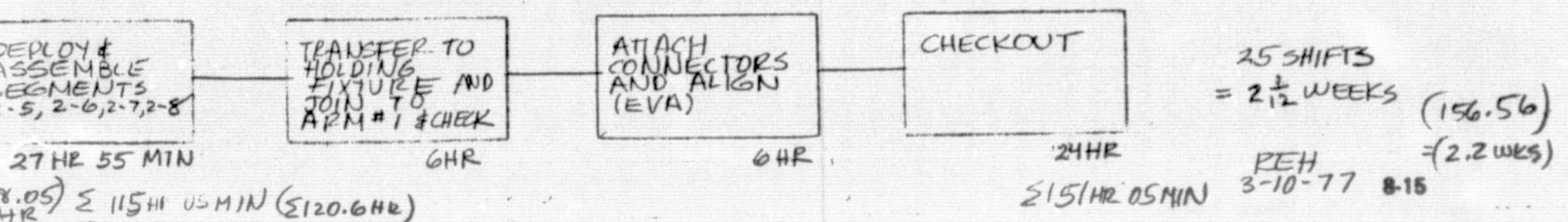
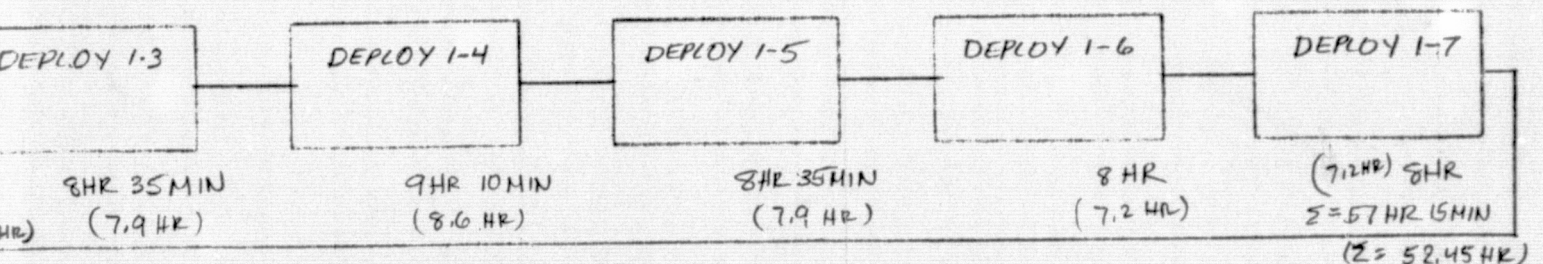
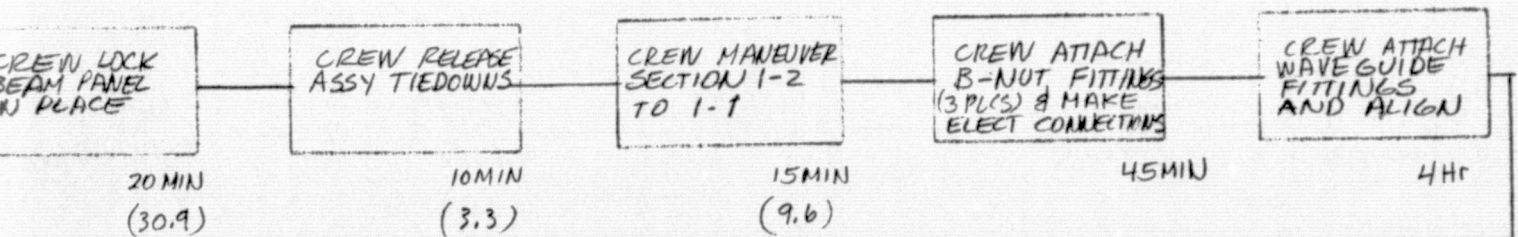
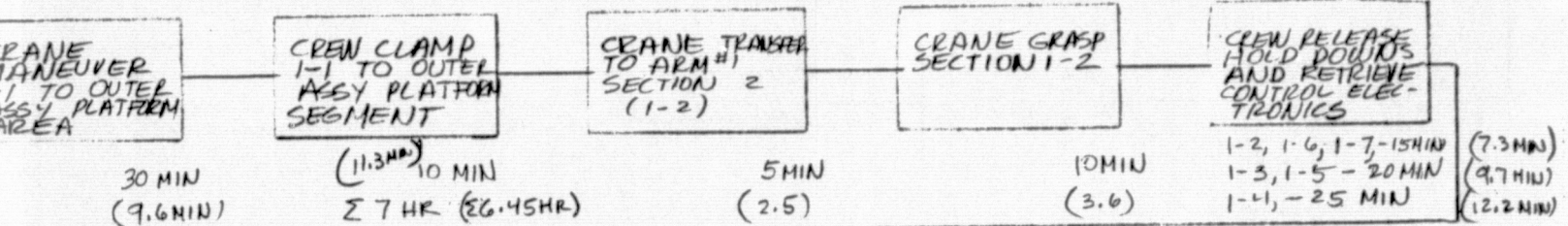
FOLDOUT FRAME 2

400 KG

EXT 10 TASKS REPEATED FOR ARM SEGMENTS 2-1 AND 2-5



NEXT 16 TASKS REPEATED FOR ARM SEGMENTS 1-3 THRU 1-7, 2-2 THRU 2-4, AND 2-6 THRU 2-8



3.2

TA-1 FABRICATION

(CHERRY PICKER UTILIZED)

TIMES ARE IN HOURS. IF IN MINUTES THEY ARE IN PARENTHESES

ORIGINAL PAGE IS OF POOR QUALITY

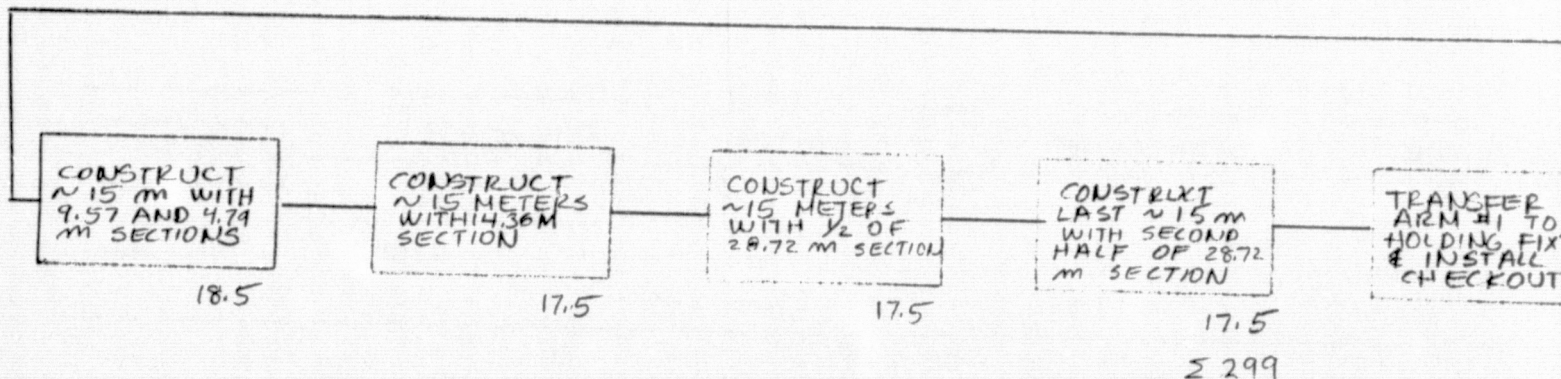
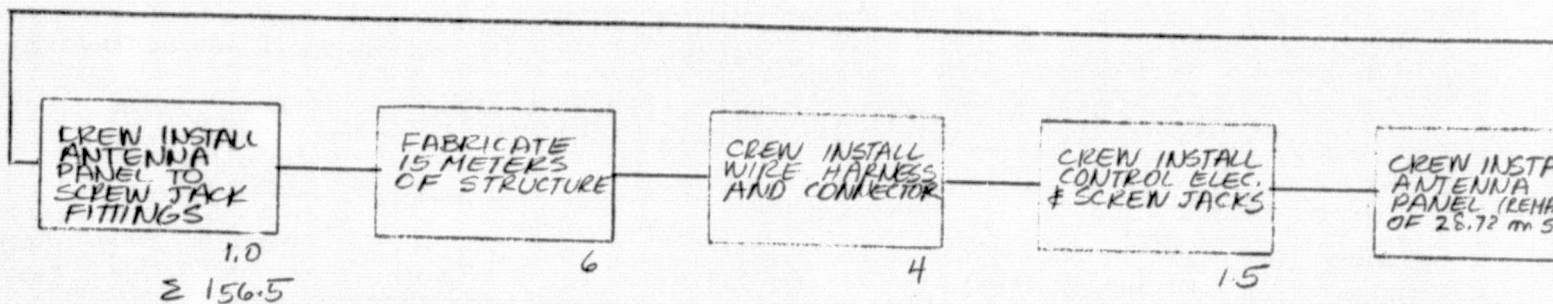
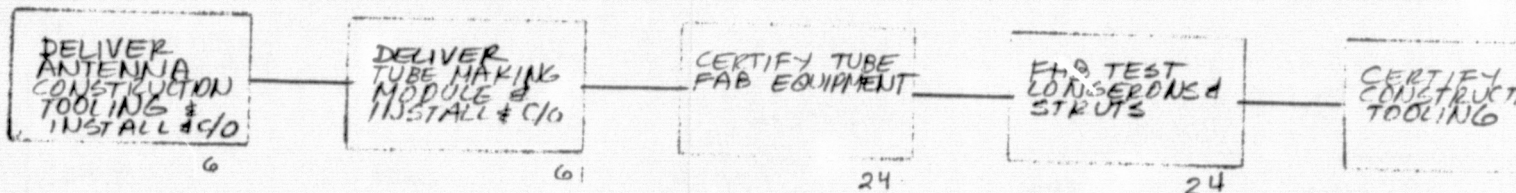
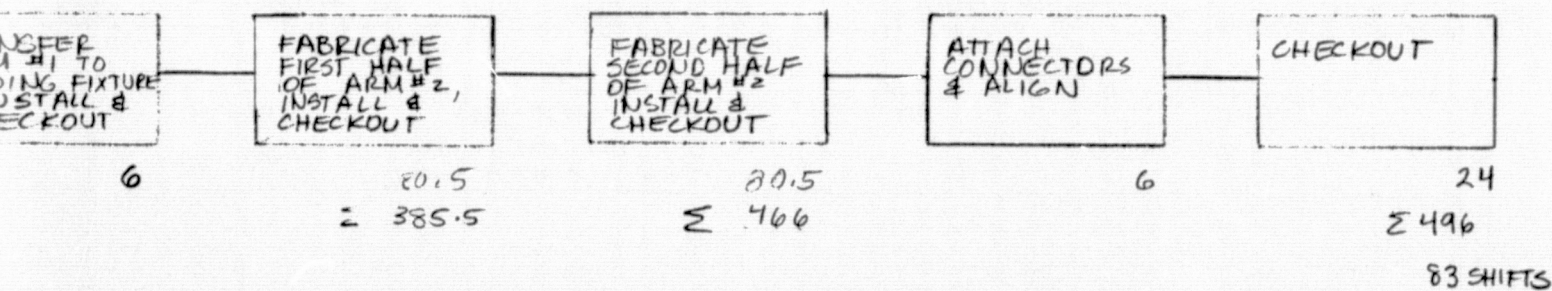
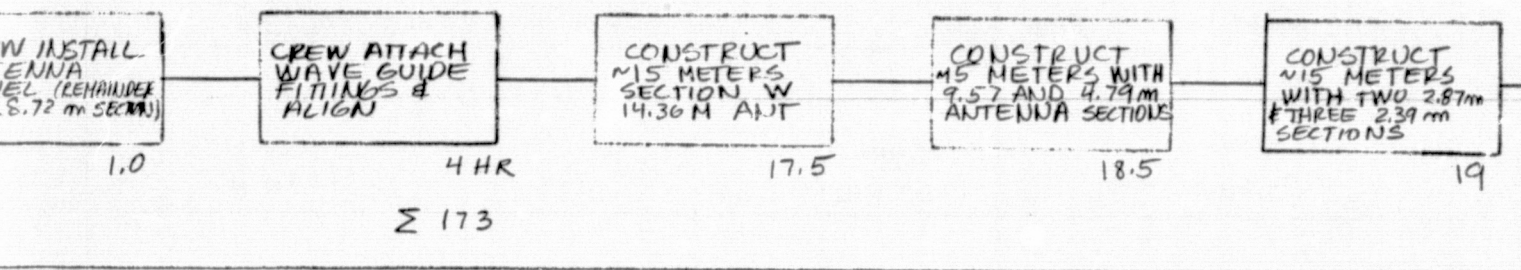
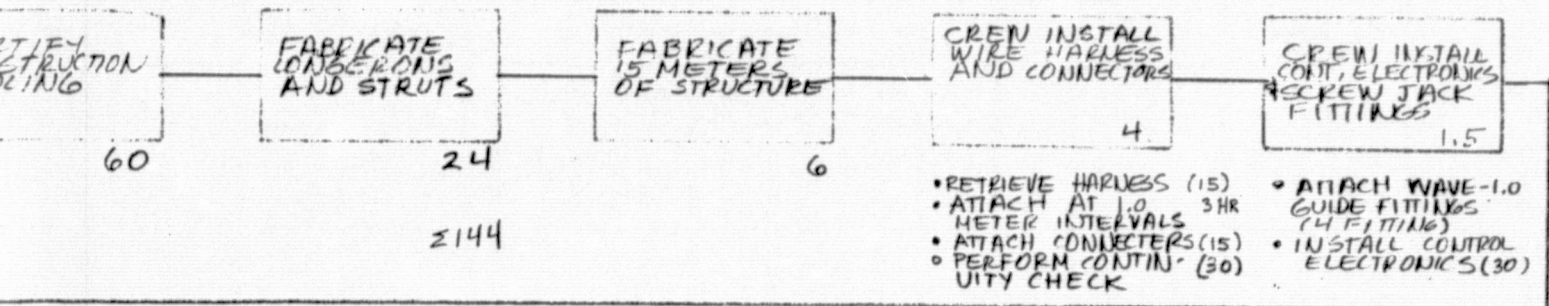


Figure 8-3. TA-1 (Sheet 2 of 6)

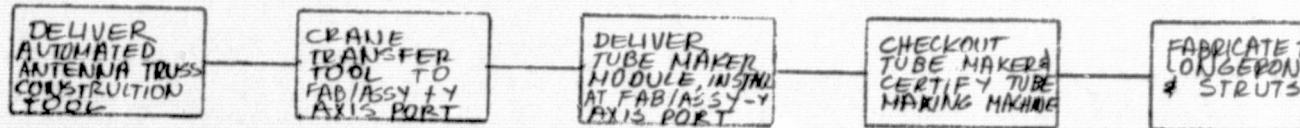


ORIGINAL PAGE 6
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FOLDOUT FRAME 4

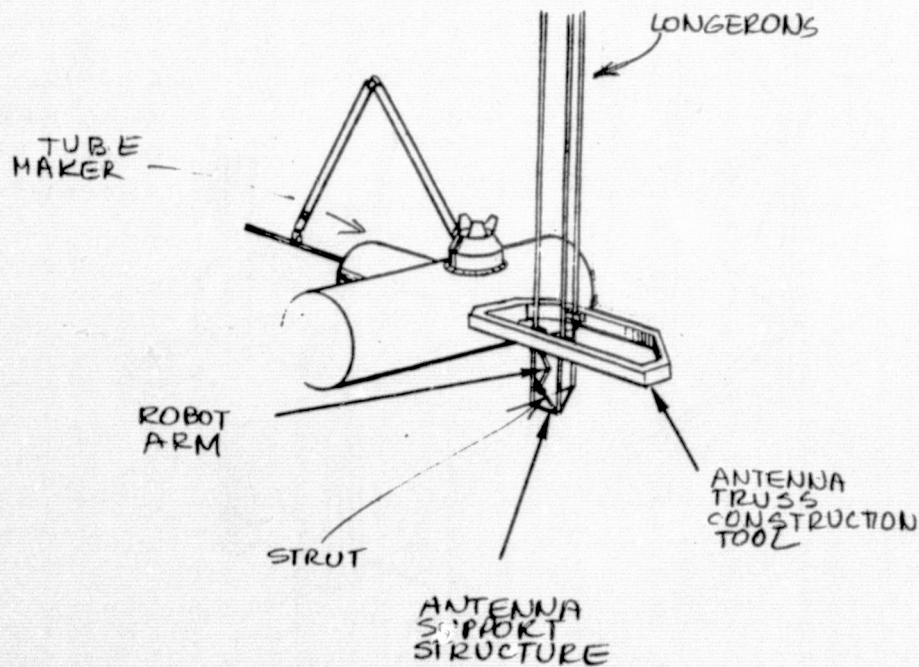
REH 3-10-77

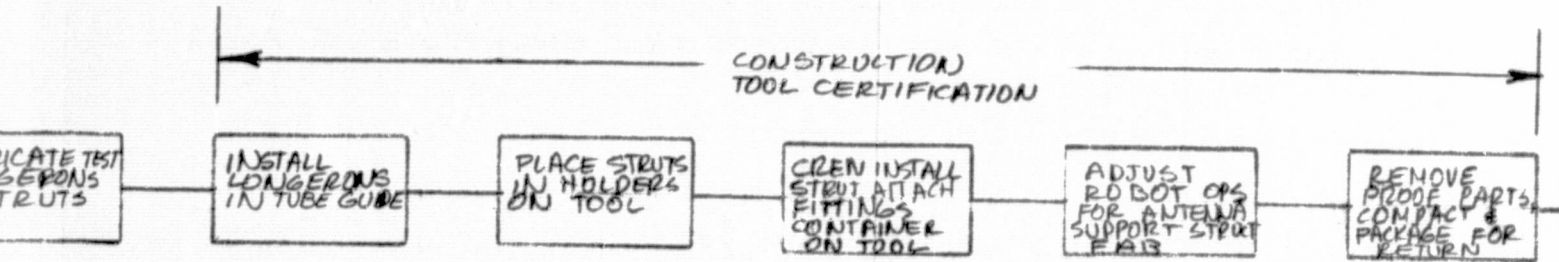
3,2,1 SPS TEST ARTICLE 1 - CONSTRUCTION



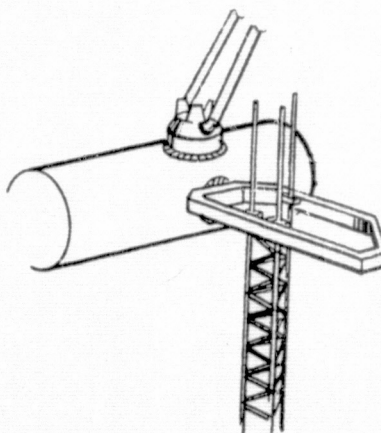
ORIGINAL PAGE IS
OF POOR QUALITY

INSTALL CONSTRUCTION
TOOLING





BUILD PROOF PARTS

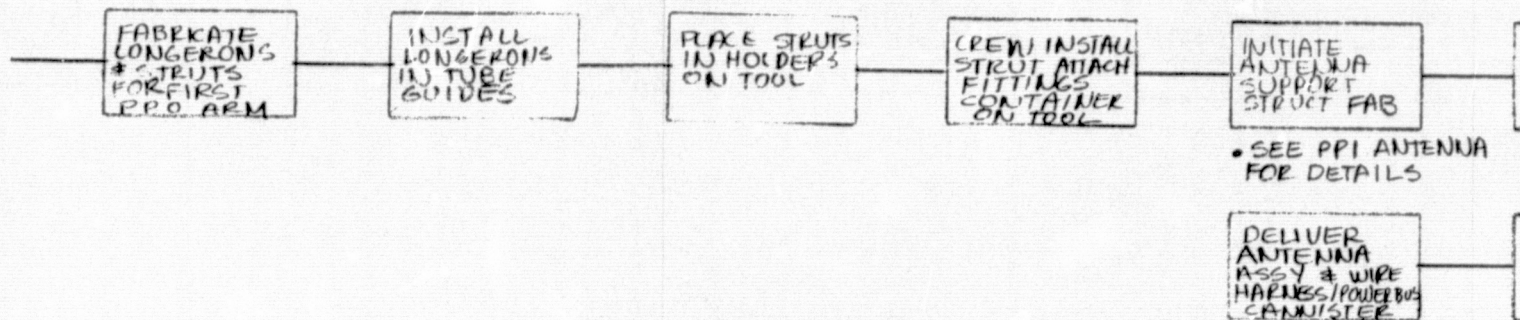


BOLDOUT FRAME

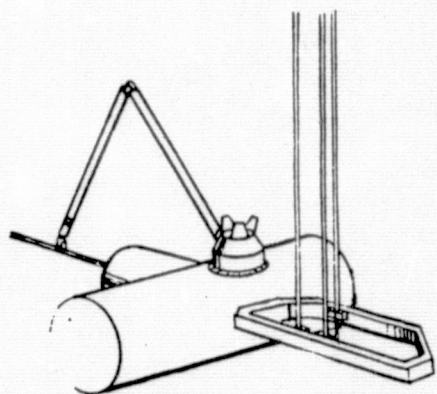
REH
12-6-76

ORIGINAL PAGE IS
OF POOR QUALITY

→ THE NEXT 14 STEPS ARE REPEATED FOR EACH ANTENNA ARM

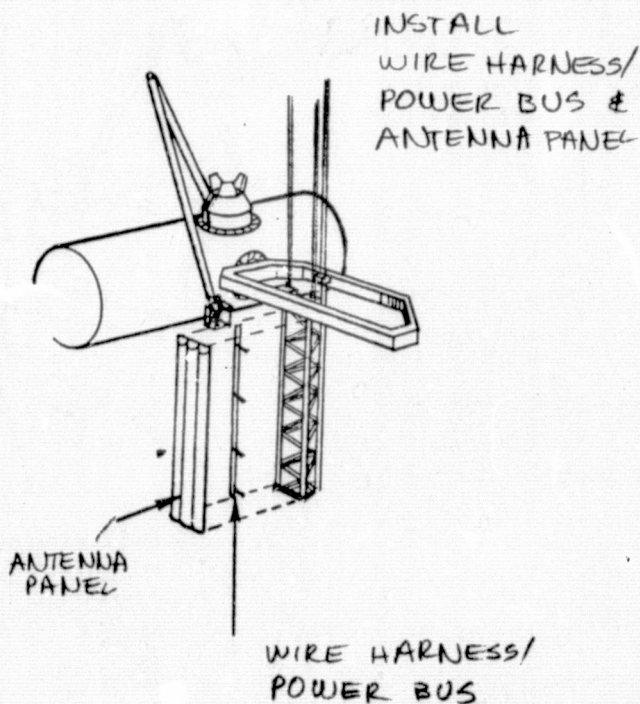
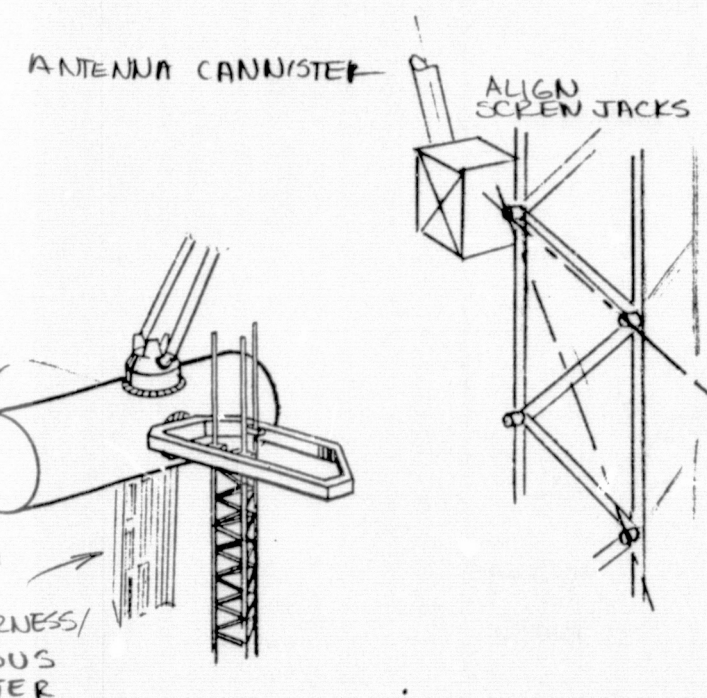
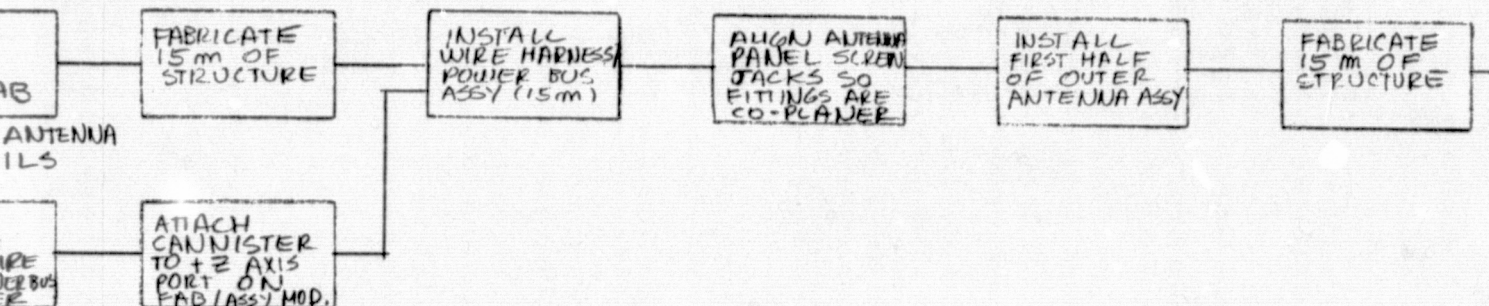


FABRICATE &
INSTALL LONGERONS
IN TOOL



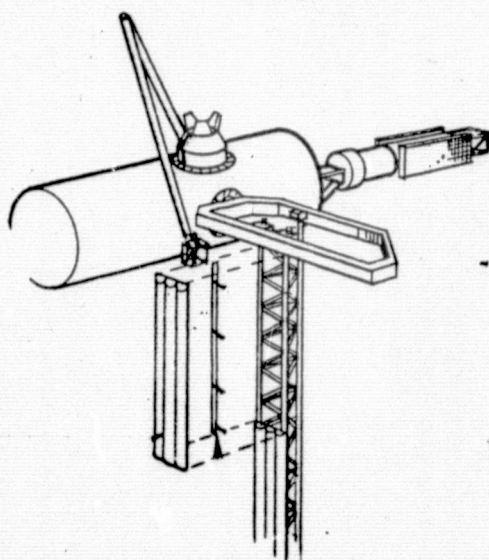
DELIVER ANTENNA

ANTENNA
ASSY &
WIRE HARNESS/
POWER BUS
CANISTER





INSTALL NEXT
WIRE HARNESS/
POWER BUS ASSY



INSTALL
WIRE HARNESS/
POWER BUS
ASSY

ALIGN
SCREW
JACKS

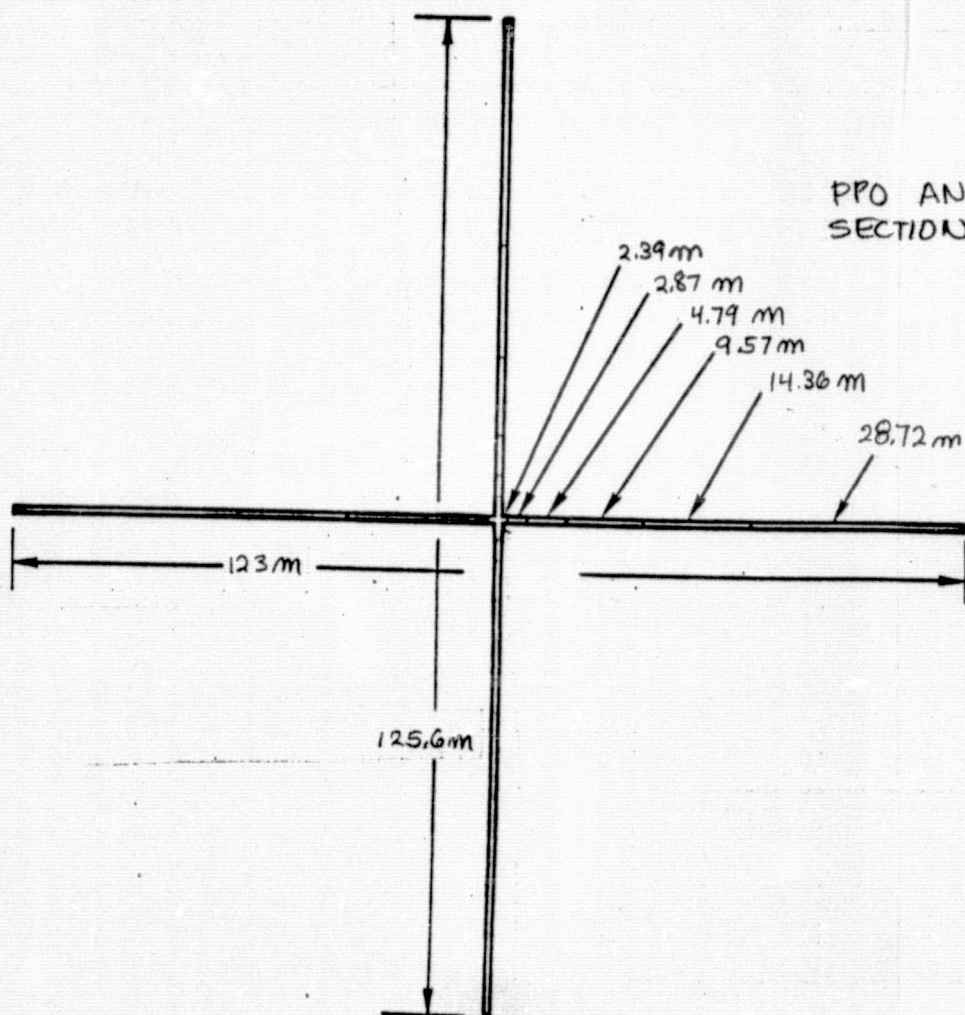
INSTALL
14.36 m
ANTENNA
ASSY

FABRICATE
FINAL 20m
OF ANT
STRUCTURE

INSTALL
WIRE HARNESS/
POWER BUS
ASSY

ALIGN
SCREW
JACKS

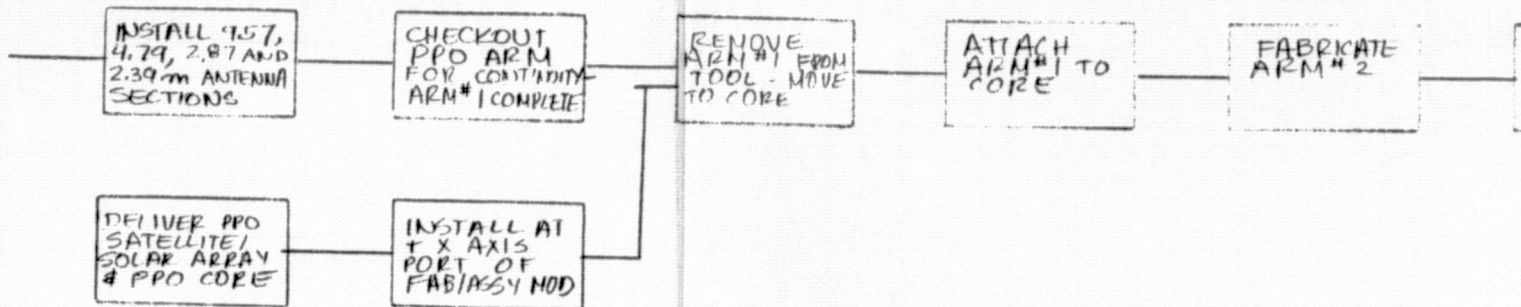
PPO ANTENNA
SECTION DIMENSIONS



FOLDOUT FRAME 2

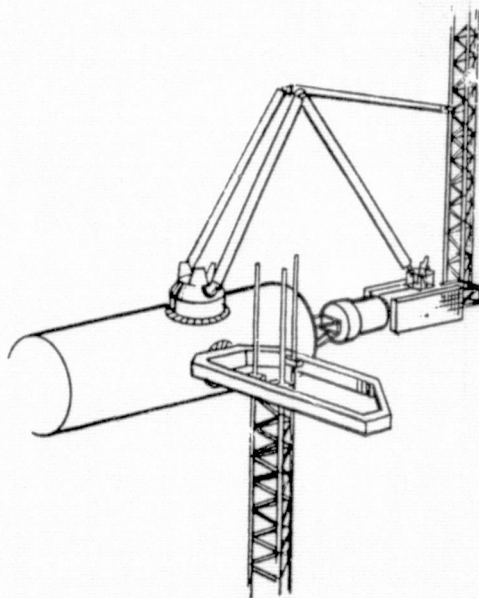
③
REH
12-6-76

ORIGINAL PAGE IS
OF POOR QUALITY



INSTALL ARM#1 IN CORE

FABRICATE
ARMS# 3



FOLDOUT FRAME 3

FABRICATE
ARM # 3

FABRICATE
ARM # 4

SHUTTLE
DELIVER BEAM
MAPPING
SATELLITE
(BMS)

ENERGIZE
ANTENNA &
PERFORM
TESTS

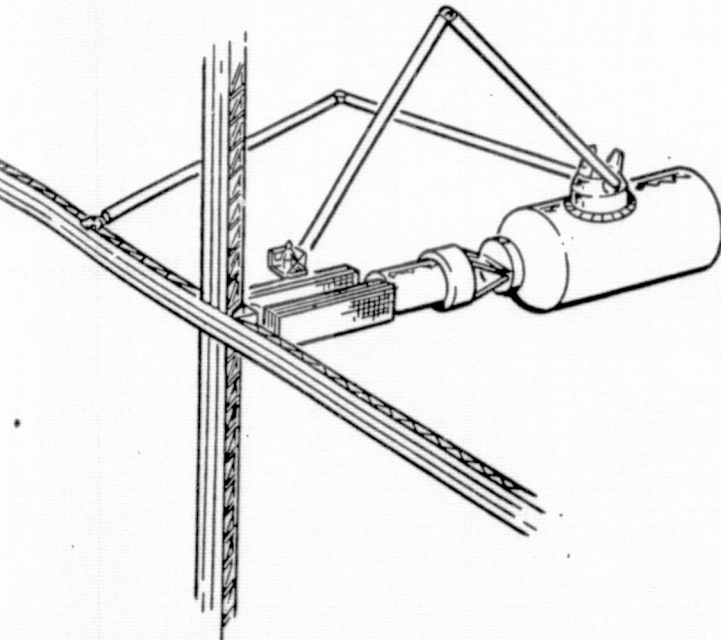
CHECK OUT
ARRAY,
DOCK IUS,
TRANSFER
TO GEO

- PLACED
120 KM
AHEAD
OF SCB
ALONG + X
AXIS

- REQUIRES
TRACKING
OF BMS

FABRICATE & INSTALL
ARMS # 3, 4 & 5

ORIGINAL PAGE IS
OF POOR QUALITY



FOLDOUT FRAME 4

(4)
REH
12-6-76

1C

4.1 TA-2 DEPLOYMENT WITH MINOR ASSEMBLY

TIMES IN HOURS & (MINUTES)

ORIGINAL PAGE IS
OF POOR QUALITY

<Revised>

5-25-77

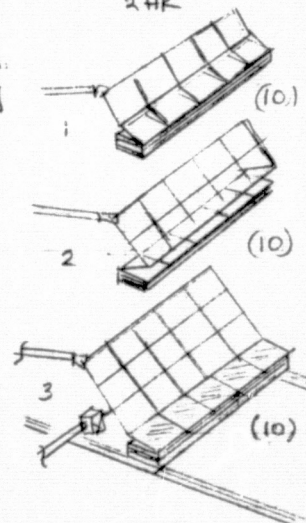
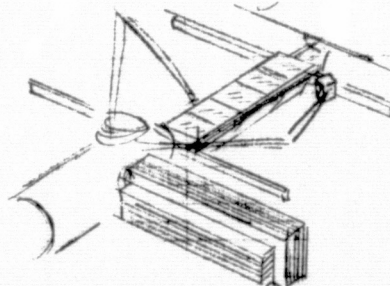
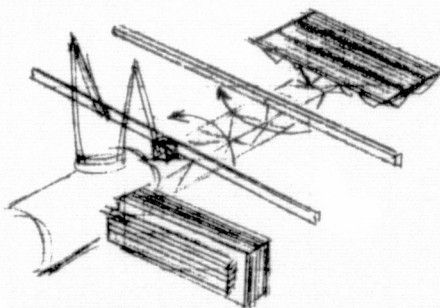
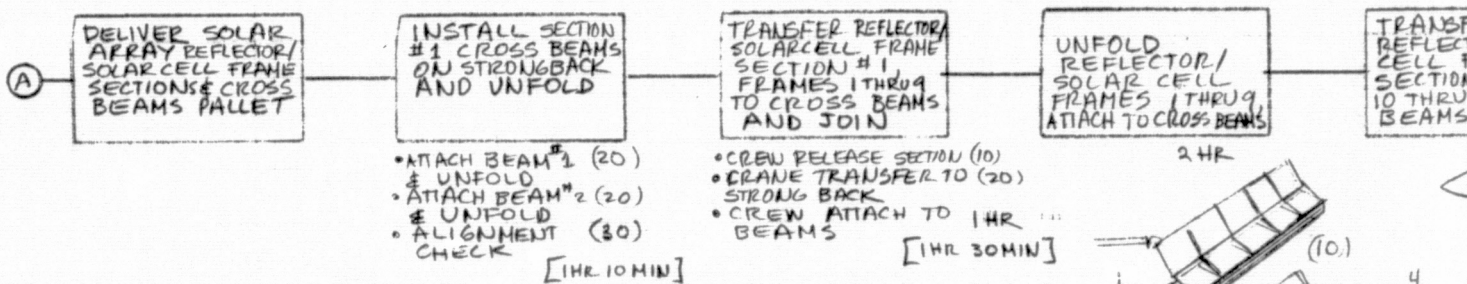
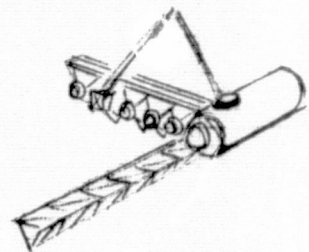
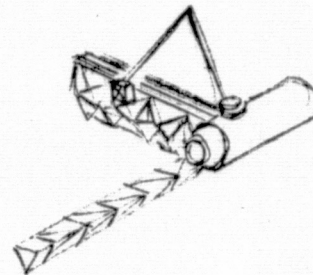
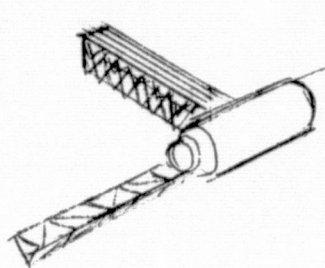
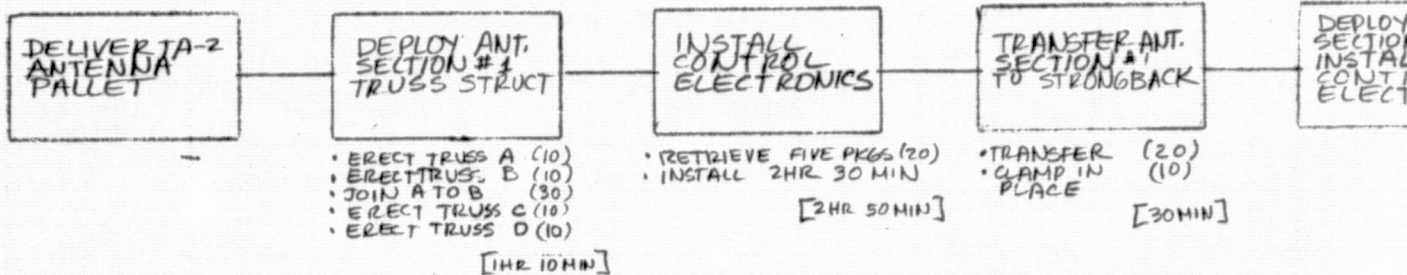


Figure 8-4. TA-2 (Sheet 1 of 10)

FOLDOUT FRAME

FOLDOUT FRAME 2

DEPLOY ANT.
SECTION #2, +
INSTALL
CONTROL
ELECTRONICS

4 HR

TRANSFER
SECTION #2
TO STRONGBACK
JOIN TO SECTION
#1 AND ALIGN

- TRANSFER (20)
- CLAMP (10)
- JOIN (15 PLCS) 3 HR 45 MIN
- JOIN CONNECTORS (15)
- ALIGN 4 HR

[8 HR 30 MIN]

DEPLOY ANT
SECTION #3
+ INSTALL
CONTROL
ELECTRONICS

4 HR

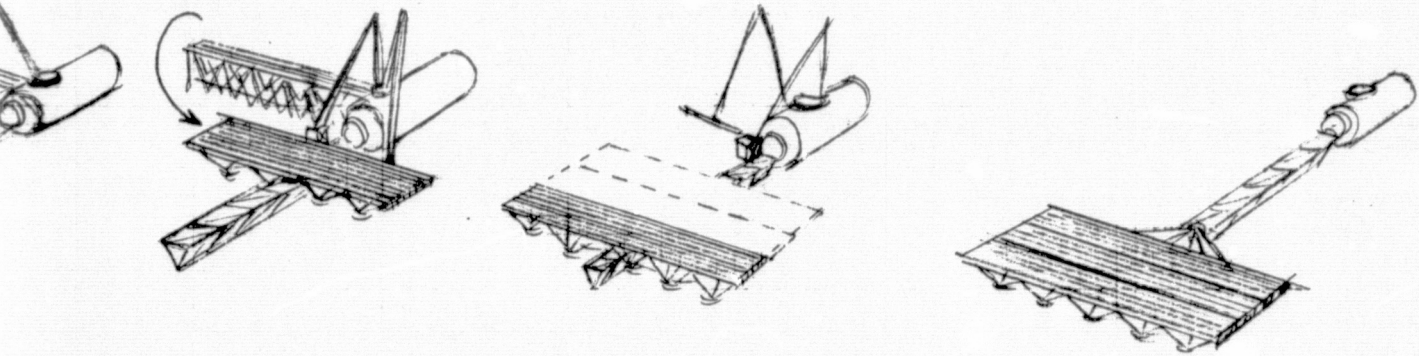
TRANSFER
SECTION #3
TO STRONGBACK
JOIN TO SECTION
#2 AND ALIGN

8 HR 30 MIN

INSTALL SOLAR
ARRAY ADAPTOR
+ GIMBAL +
CHECKOUT

- DEPLOY ADAPTOR (30)
- TRUSS
- CONNECT TO 1 HR
- ANTENNA (4 PLCS)
- ATTACH GIMBAL 1 HR
- CLAMP TO END OF STRONGBACK (10)

[2 HR 40 MIN]



TRANSFER
REFLECTOR SOLAR
CELL FRAME
SECTION #1, FRAMES
10 THRU 15 TO CROSS
BEAMS + JOIN

3 HR 30 MIN
(1 HR 30 MIN)

UNFOLD
REFLECTOR
FRAMES
10 THRU 15 AND
ATTACH TO END
OF CROSS BEAMS

1 HR 30 MIN

INSTALL GIMBAL
ADAPTOR AND
ATTACH SECTION
#1 TO ANTENNA
ASSEMBLY

- INSTALL STRUTS (1 HR)
- (4 PLCS)
- RELEASE GIMBAL (10)
- ATTACH GIMBAL 1 HR

[2 HR 10 MIN]

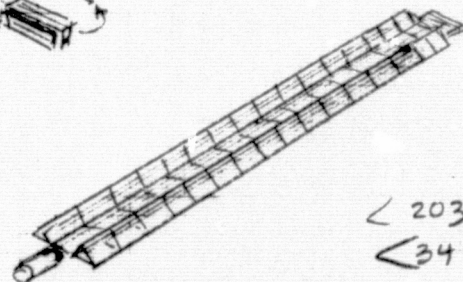
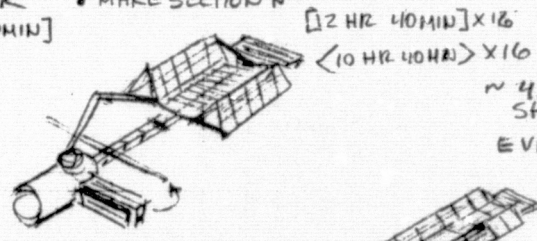
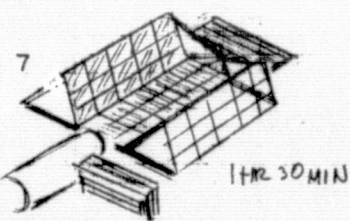
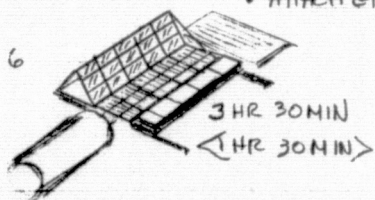
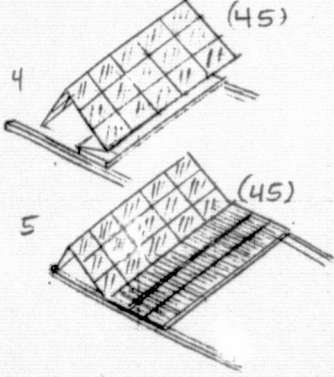
MOVE ANTENNA
SECTION #1
ASSEMBLY
OUTWARD AND
CONSTRUCT SECTION
#2 - 17

- CREW RELEASE BRAKE (10)
- TRANSLATE SECT #1 (30)
- CREW CLAMP (10)
- MAKE SECTION "N"

[2 HR 40 MIN] X 16
[10 HR 40 MIN] X 16

CHECKOUT
TA-2

12



~ 42
SHIFTS
EVA 34 SHIFTS

< 203 >
< 34 SHIFTS >

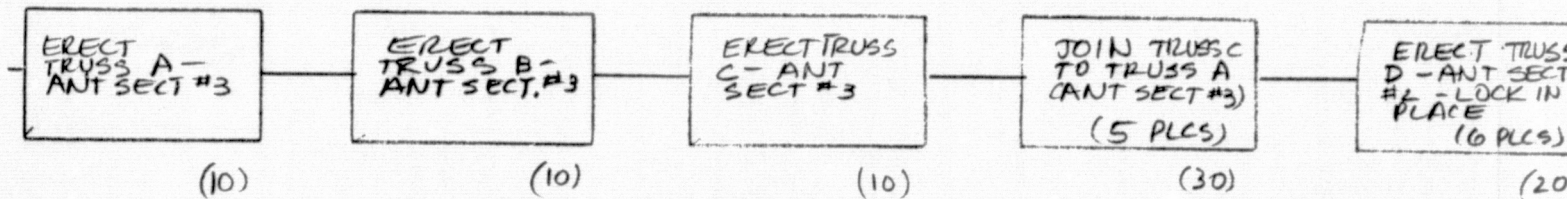
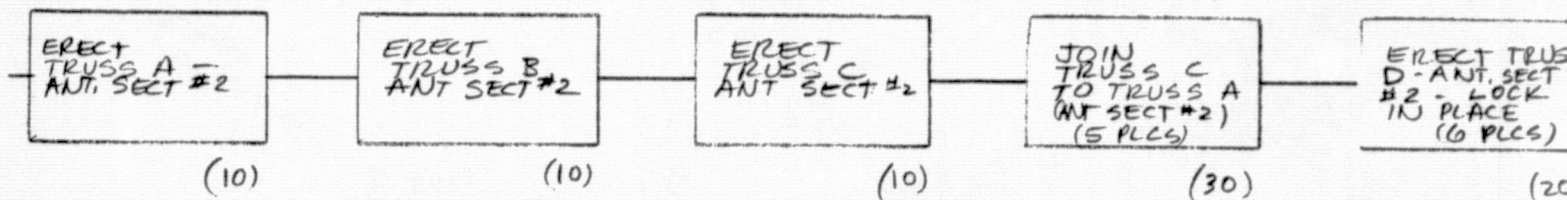
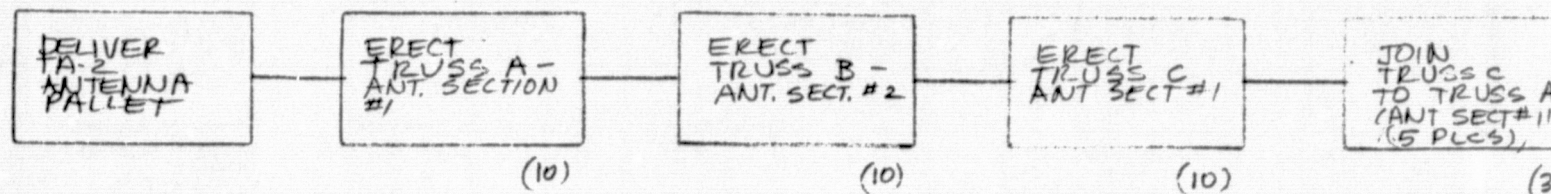
SOLAR ARRAY PORTION
ONLY 456 KW = 29 SHIFTS
250 KW = 18 SHIFTS

REH
3-11-77

4.1.1

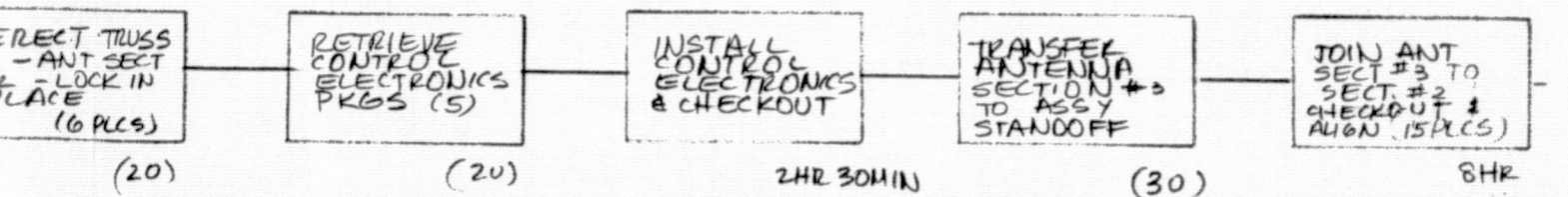
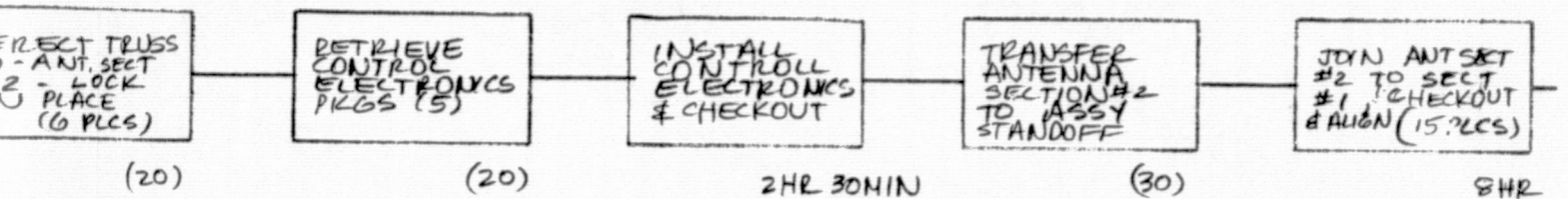
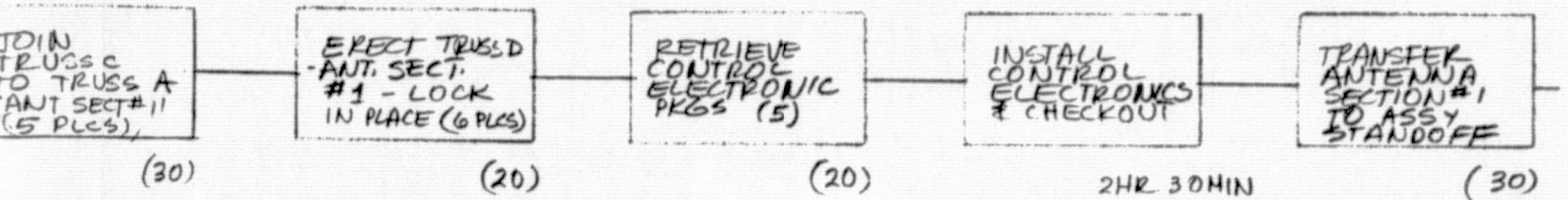
TA-2 ANTENNA DEPLOYMENT HOURS (MIN)

ORIGINAL PAGE IS
OF POOR QUALITY



FOLDOUT FRAME

ORIGINAL PAGE IS
OF POOR QUALITY



$\Sigma = 30 \text{ HRS}$
 $\sim 2\frac{1}{2} \text{ DAYS}$

EXPLOSION FR 4

REH 5-6-77

4.2 TA-2 FABRICATION (CHERRY PICKER UTILIZED)

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TIME HRS (MIN)

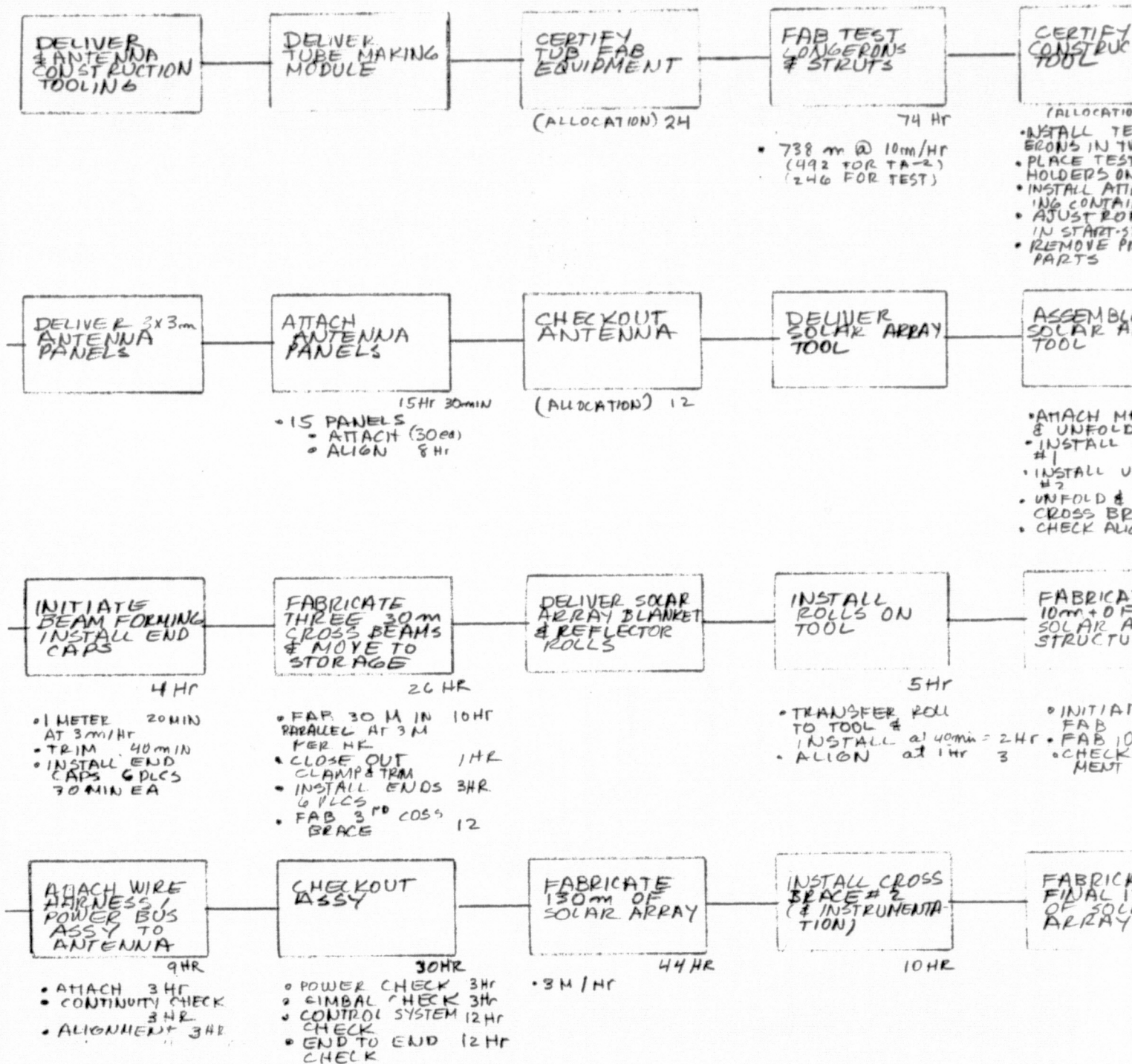


Figure 8-4. TA-2 (Sheet 3 of 10)

FOLDOUT FRAME

ORIGINAL PAGE IS
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CERTIFY CONSTRUCTION TOOL

(ALLOCATION) 60 HR
• INSTALL TEST LONG-
FRONS IN TUBE GUIDE
• PLACE TEST STRUTS
HOLDERS ON TOOL
• INSTALL ATTACH FIT-
ING CONTAINER
• ADJUST ROBOT OPS
IN START-STOP FAB
• REMOVE PROOF
PARTS

FABRICATE ANTENNA SUPPORT STRUCTURE

24 HR
• INSTALL LONGERONS
IN TUBE GUIDE 3 HR
• INSTALL STRUT &
ATTACH FITTING
HOLDERS 3 HR
• FAB TRUSS AT
2m / HR - 15 HR
• TRIM & CHECK
FITTINGS 3 HR

INSTALL & ALIGN SCREW JACK FITTINGS

30 HR
• 4.5 FITTINGS
• INSTALL (15) ea.
• ALIGN (25) ea.

INSTALL ANTENNA ATTACH & GIMBAL

6 HR
• ASSEMBLE 2
• ATTACH TRUSS
• ATTACH TO
ANTENNA
FRAME 1
• INSTALL
GIMBAL 1
• ALIGN 1

INSTALL WIRE HARNESS & CONTROL ELECTRONICS

35 HR
• WIRE HARNESS
ATTACH (120 PLS) @ 10
• INSTALL CONT.
ELECTRONICS
ASSEMBLIES 30
(15) - 15

ASSEMBLE SOLAR ARRAY TOOL

19 HR
• ATTACH MAIN BEAM
& UNFOLD 3 HR
• INSTALL UPRIGHT 3 HR
• INSTALL UPRIGHT 3 HR
• UNFOLD & INSTALL 4 HR
• CROSS BRACE
CHECK ALIGNMENT 6 HR

DELIVER BEAM CAP MAKERS & EQUIPMENT

INSTALL BEAM CAP MAKERS & STRUT FASTENERS

12 HR
• TRANSFER CAP
MAKER TO
FRAME &
INSTALL 1 Hr ea - 6
• ALIGN 1 Hr ea - 6

INSTALL TRUSS MAGAZINES ALIGN TOOL & ROBOT EQUIPMENT

4 HR
• TRANSFER
TO FRAME
& INSTALL
1 Hr ea 2
• ALIGN
1 Hr ea 2

CERTIFY FABRICATION FACILITY

(ALLOCATION)
132 HRs

FABRICATE 10m+ OF SOLAR ARRAY STRUCTURE

8 HR
• INITIATE STRUCT
FAB 4 HR
• FAB 10m 3 HR 20 MIN
• CHECK ALIGN-
MENT 40 MIN

INSTALL CROSS BRACE #1

10 HR
• TRANSFER FROM
STORAGE 1 HR
• CLAMP IN PLACE 30 MIN
• MAKE ATTACHMENTS
(8 @ 1 Hr) 8
• CHECK ALIGNMENT
30 MIN

INSTALL ANTENNA ATTACH FITTING

5 HR

INSTALL ANTENNA & ALIGN

9 HR
• TRANSFER
ANTENNA &
ATTACH TO GIMBAL 3 HR
• ALIGN & CHECKOUT 6

UNROLL BLANKET & REFLECTOR ROLLS & INSTALL

20 HR
• UNROLL REFLECTOR
#1 INSTALL
CORNER FITTINGS 1 HR
• INSTALL TENSION
FITTINGS 20 PLS 5 HR
• REPEAT FOR
REFLECTOR NO. 2 6 HR
• UNROLL & ATTACH BLANKET
• ATTACH FITTINGS 4 HR
• ADJUST ROBOTS 12 HR

FABRICATE FINAL 125m+ OF SOLAR ARRAY

42 HR

INSTALL CROSS BRACE #3

10 HR

CLAMP ASSEMBLY

3 HR

CUT, TRIM & CLOSEOUT STRUCTURE

4 HR

PERFORM CHECKOUT

12 HR

SOLAR ARRAY FAB ONLY
450 KW = 48 SHIFTS
250 KW = 36 SHIFTS

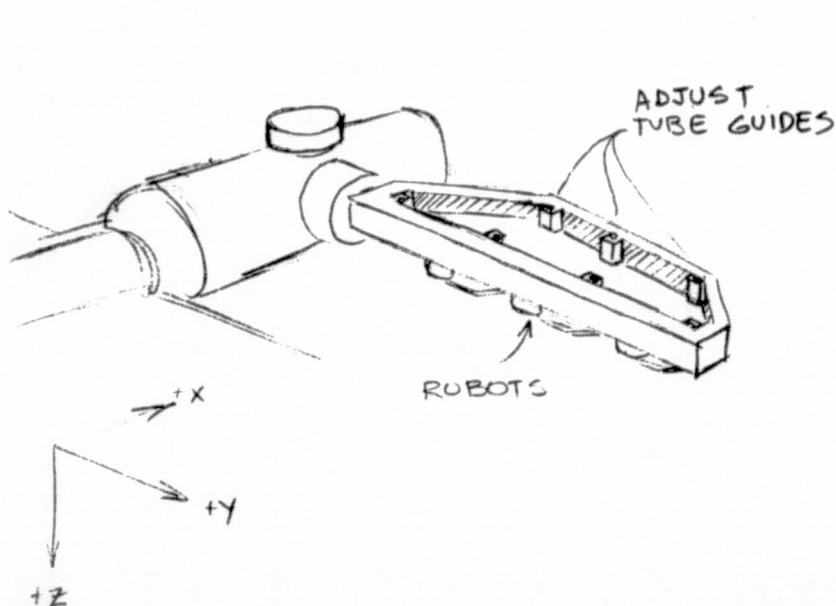
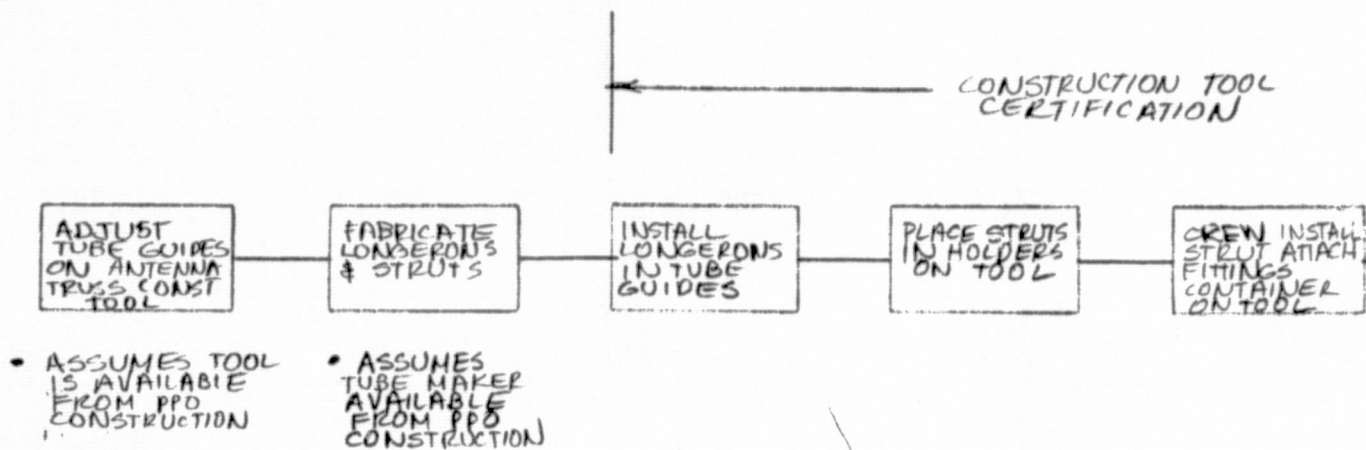
2 701.5

117 SHIFTS
9 3/4 WEEKS

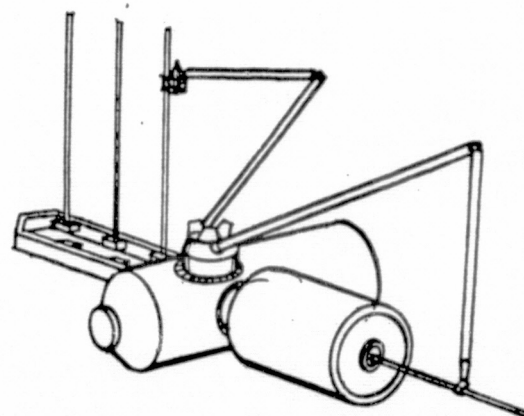
REH
3-14-77
8-23

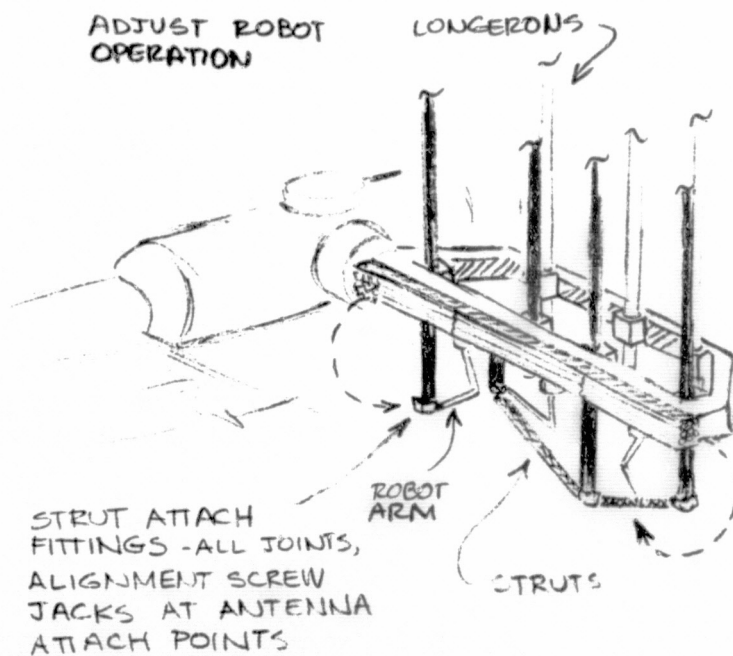
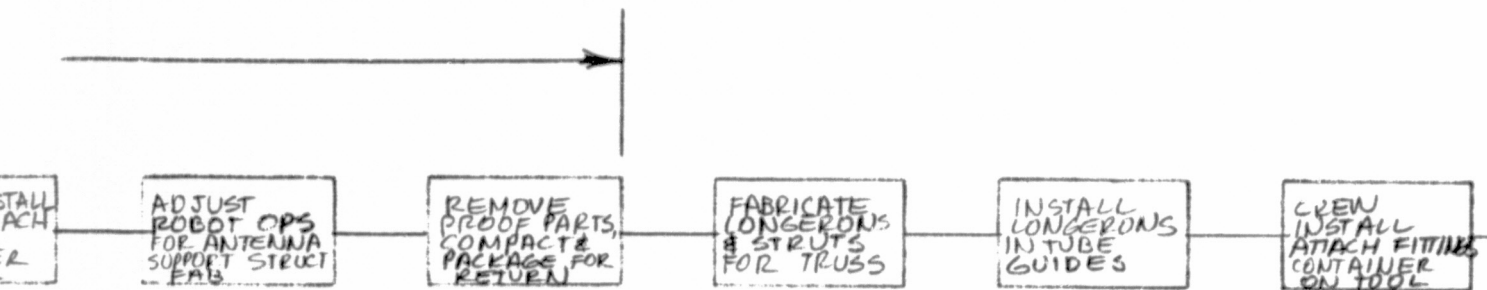
OLDOUT FRAM 2

4.2.1 SPS TEST ARTICLE-2 CONSTRUCTION



FAB LONGERONS - INSTALL IN TUBE GUIDES





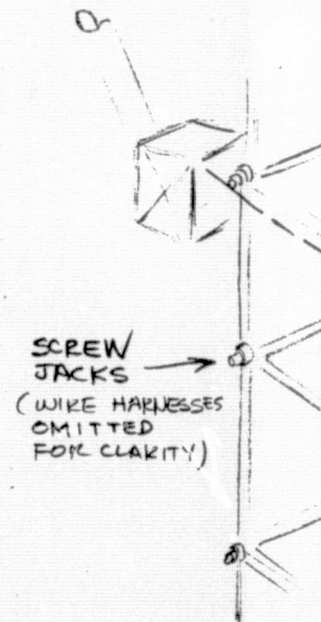
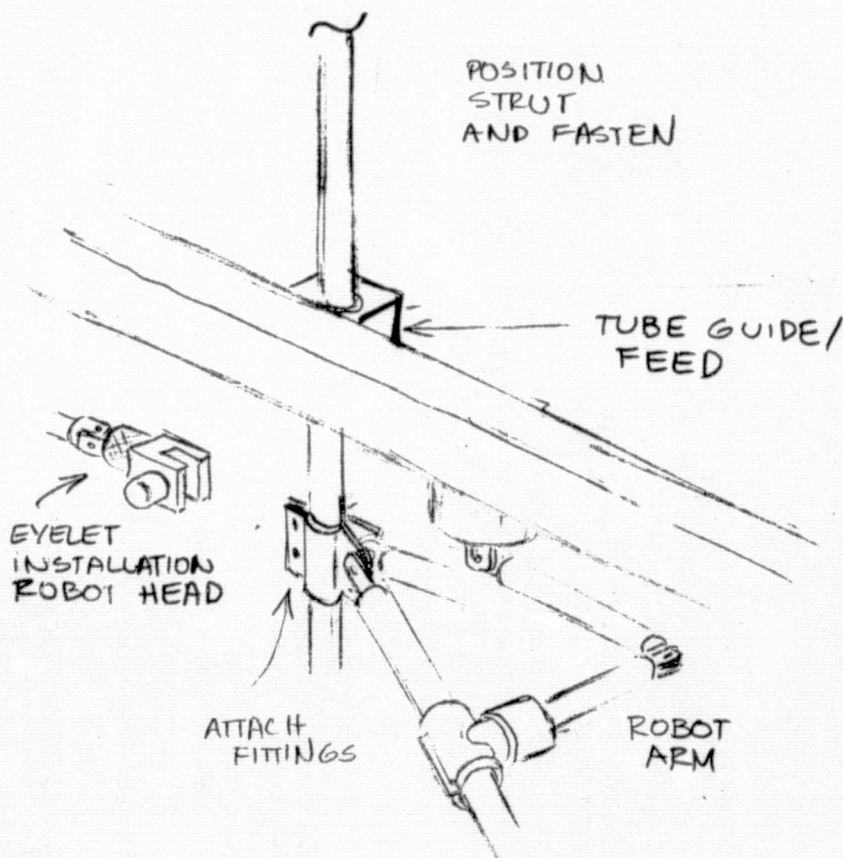
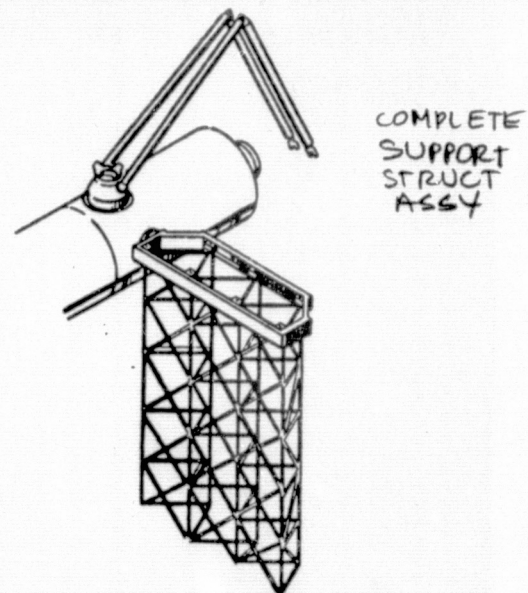
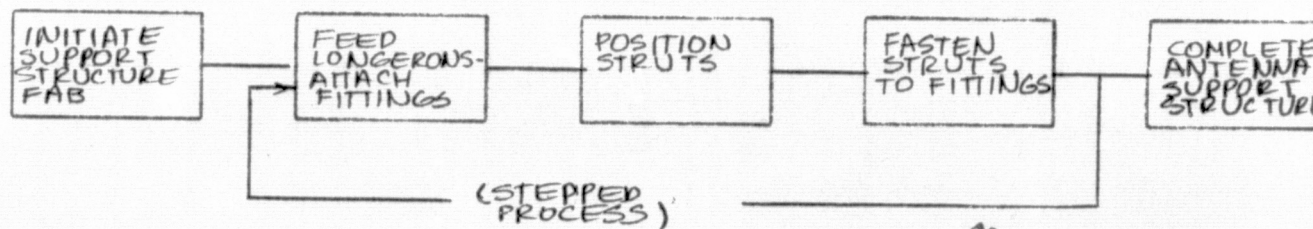
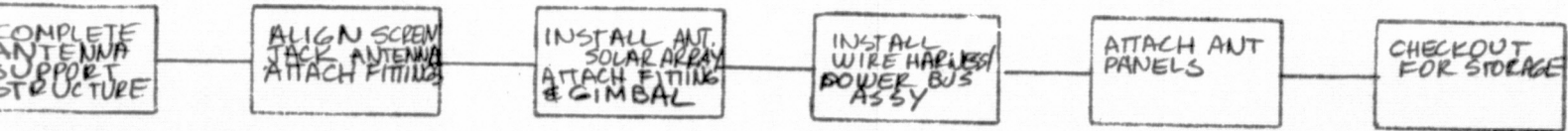


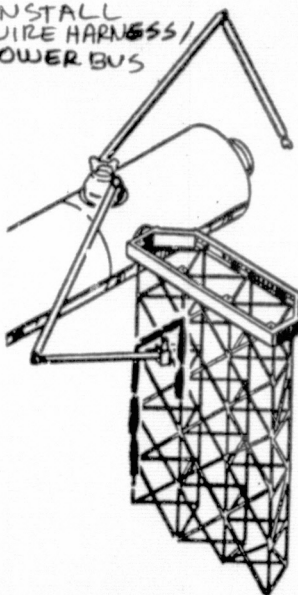
Figure 8-4. TA-2 (Sheet 5 of 10)

FOLDOUT FRAME

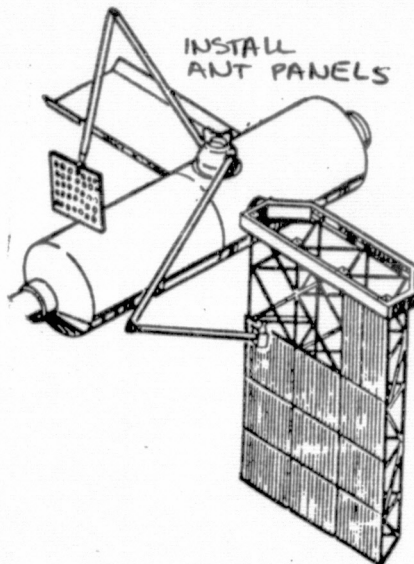


COMPLETE
SUPPORT
STRUCTURE
ASSY

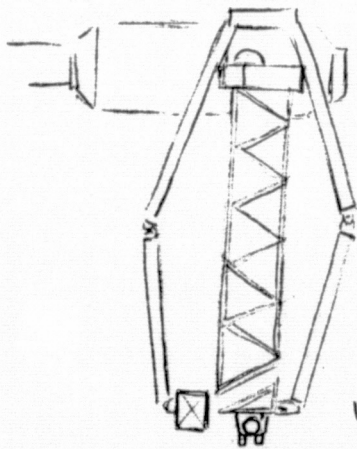
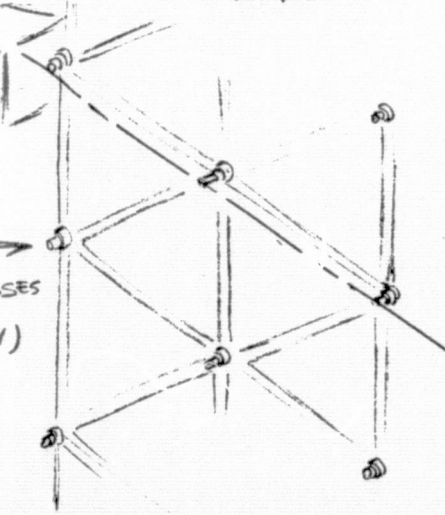
INSTALL WIRE HARNESS/
POWER BUS



INSTALL ANT PANELS



ALIGN SCREW
JACKS

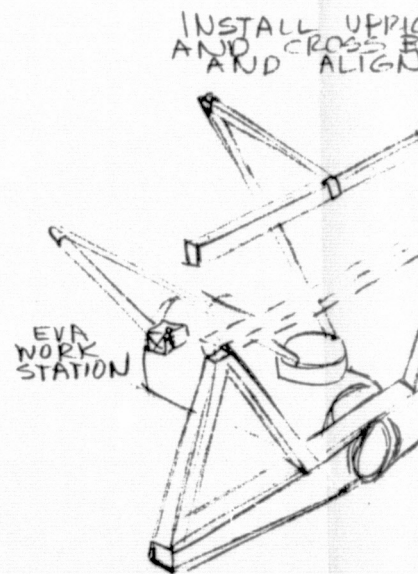
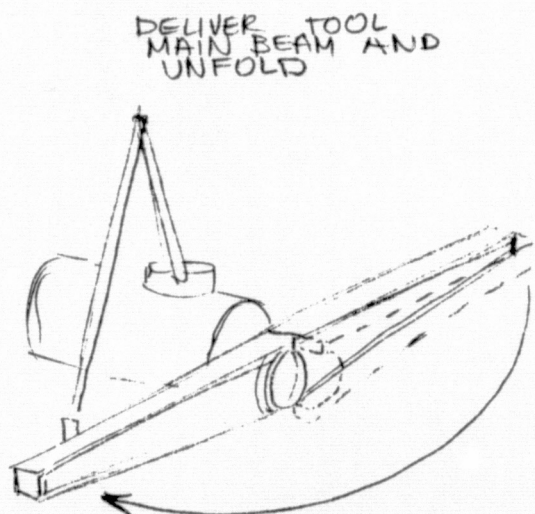
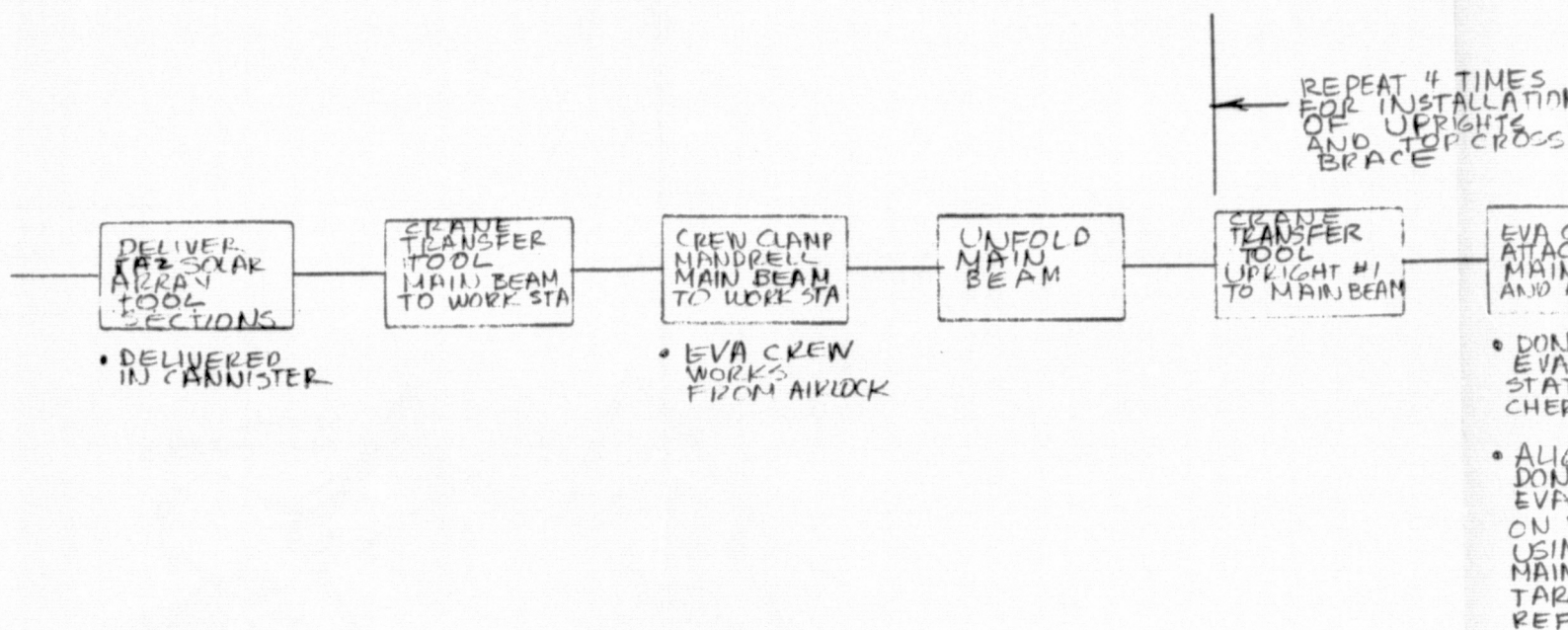


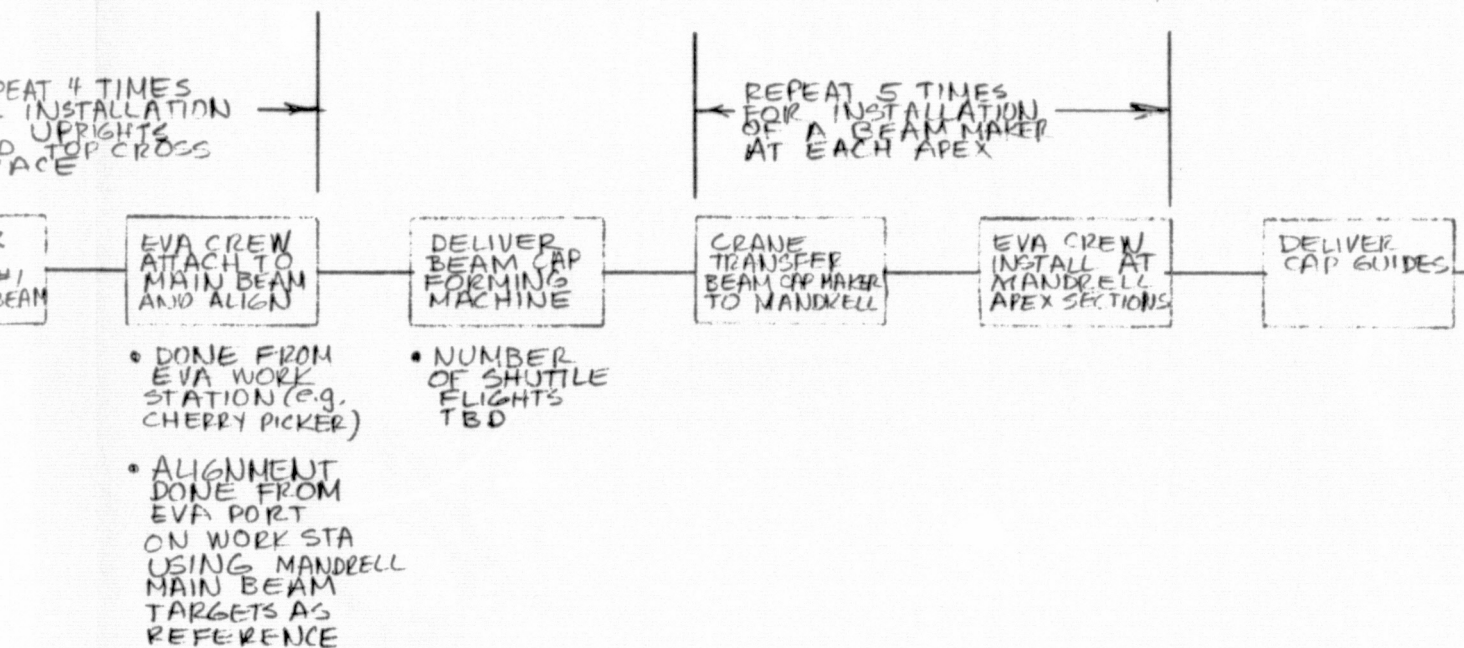
INSTALL
ANTENNA
ATTACH
FITTING
& GIMBAL

FOURTH FRAME 2

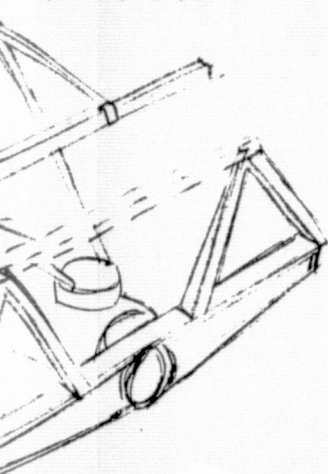
②
KEH
12-6-76
8-26

TA-2 SOLAR ARRAY & FINAL ASSEMBLY

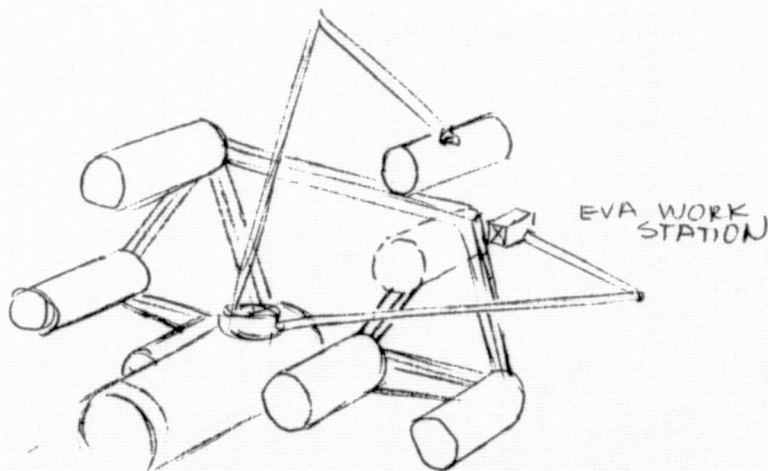




STALL UPRIGHTS
D. TOP CROSS
FACE AND ALIGN



INSTALL BEAM CAP
MAKERS



FOLDOUT FRAME L

①

REH

11-1-76

12-6 76

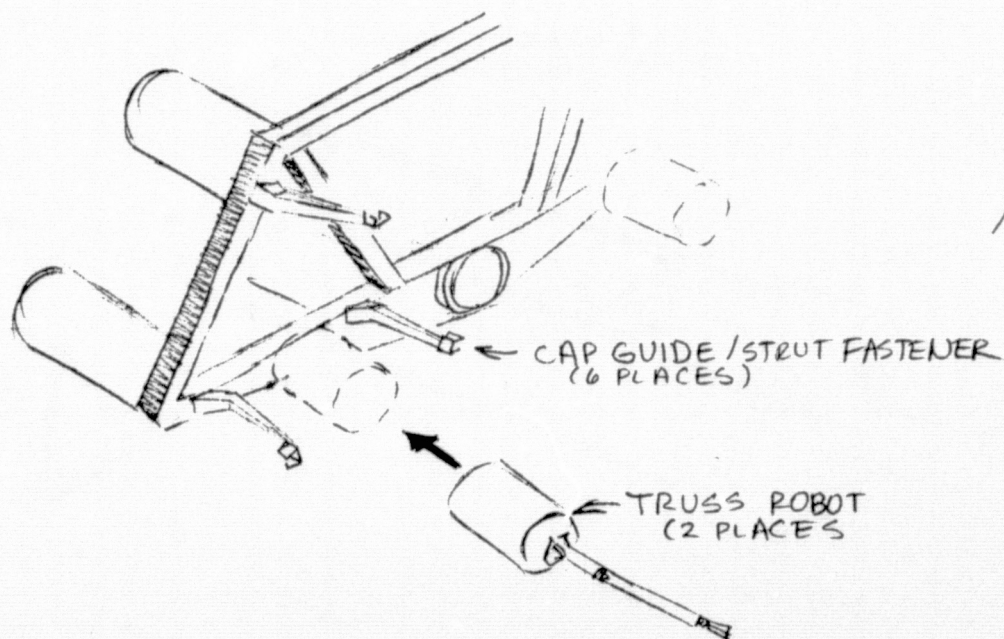
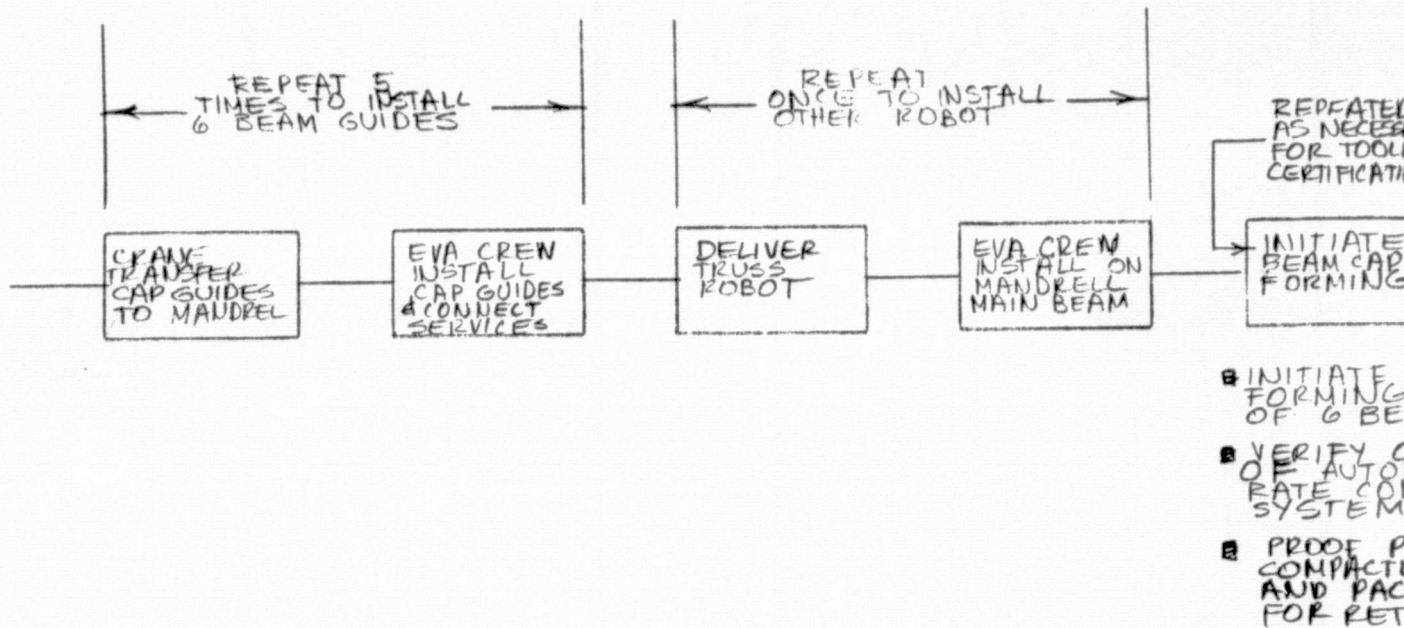


Figure 8-4. TA-2 (Sheet 7 of 10)

FOLDOUT FRAME

REPEATED
AS NECESSARY
FOR TOOLING
CERTIFICATION

INITIATE
BEAM CAP
FORMING

INSTALL
END CAPS

FABRICATE
(TBD) M.
OF BEAM
CAPS

CRANE
DELIVER A
END FRAME
STRUTS TO END
OF BEAMS

EVA CREW
INSTALL
END FRAME
STRUTS

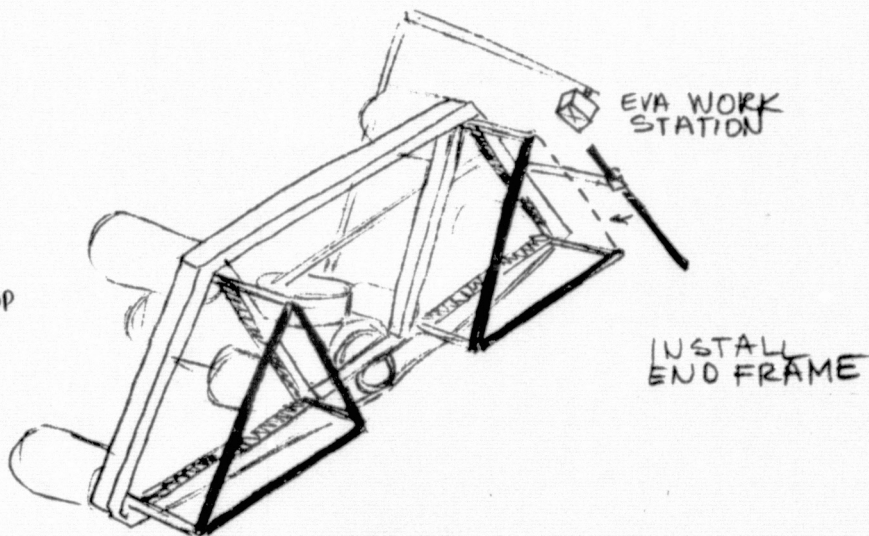
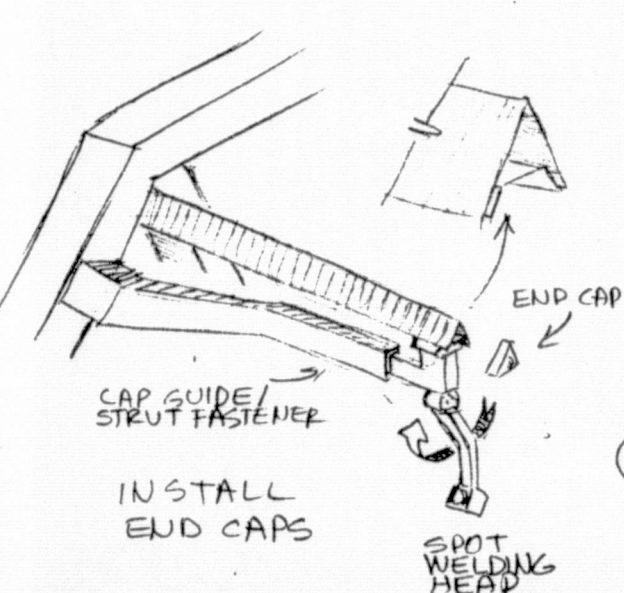
DELIVER
TRUSS
MAGAZINES

INITIATE
FORMING
F G BEAMS

VERIFY OPERATION
F AUTOMATIC
CONTROL
SYSTEM

PROOF PARTS
COMPACTED
AND PACKAGED
FOR RETURN

■ DONE
FROM
EVA WORK
STATION



FOLDOUT FRAME 2

②

REH

11-1-76

12-6-76

8-27

REPEAT PROCESS
FOR EACH
CROSS BRACE
TRUSS

CRANE
TRANSFER
TRUSS
MAGAZINES

EVA CREW
INSTALL
TRUSS
MAGAZINES (2)

FABRICATE
TBD METERS
OF BEAM
CAPS

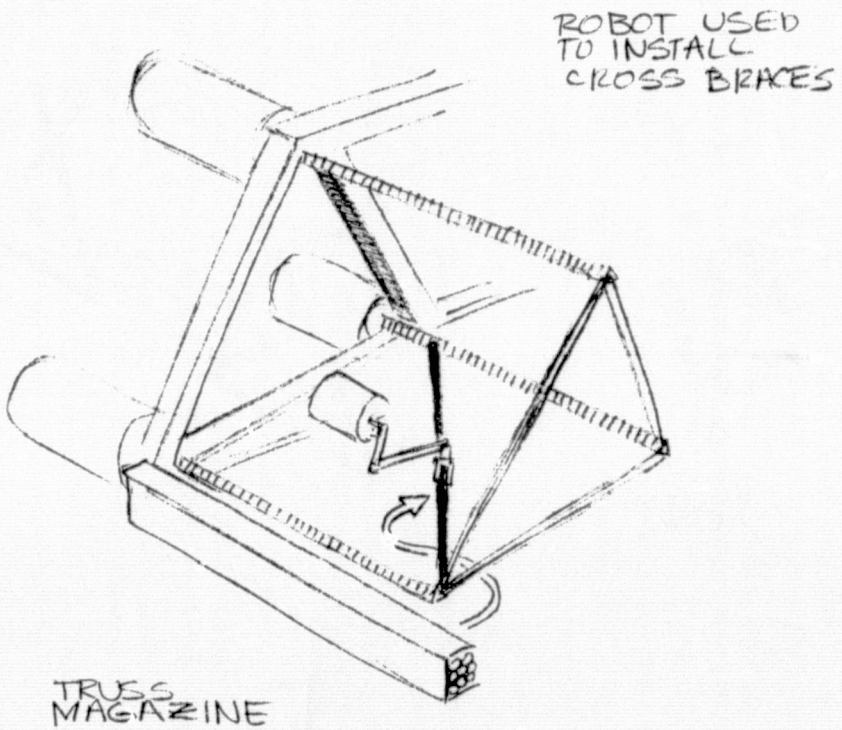
ROBOT
PLACE
CROSS BRACE
TRUSS AGAINST
BEAM CAP

FASTEN
TRUSS TO
END-CAP

CL
F
8
30

■ CROSS BRACE
TRUSS FIXED
TO BEAM CAPS
BY SPRING
CLIP

■ ATTACHMENT
TECHNIQUE
TBD (E.G.
SPOT WELDING)



FOLDOUT FRAME 3

ENT
E
6.
LDING)

CONTINUE
FABRICATION
#2 ASSY UNTIL
30 M COMPLETE

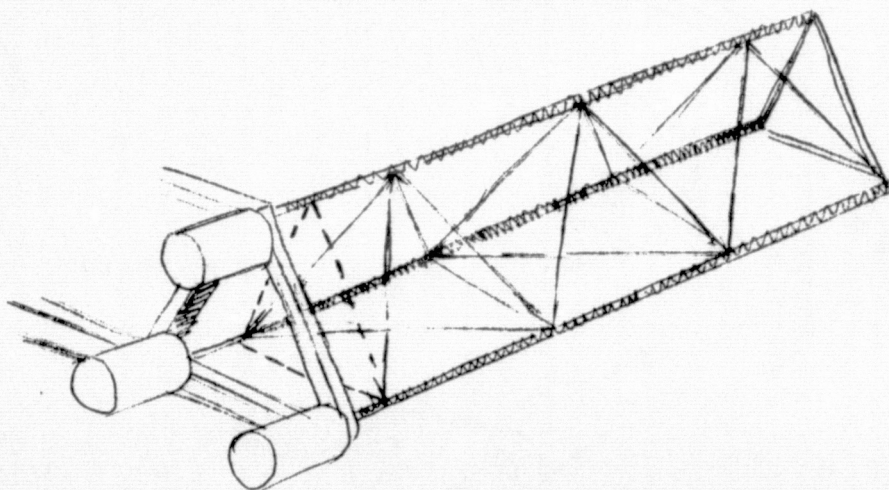
FABRICATE
AN ADDITIONAL
TED M OF
BEAM

CLAMP Δ
BEAM #1
WITH CRANE
ARM

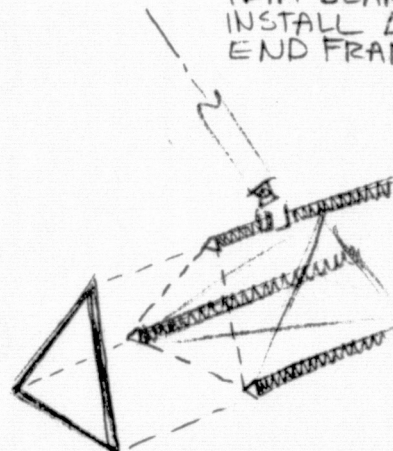
EVA CREW
CUT & TRIM
BEAM CAPS

EVA CREW
INSTALL
END CAPS
& Δ END
FRAME STRUTS

30 M LENGTH CROSS
BEAM FABRICATED



CUT AND
TRIM BEAM,
INSTALL Δ
END FRAME



③

REH

11-1-76

12-6-76

COLDOUT FRAME 4

FABRICATION & ASSEMBLY
STEPS ARE ESSENTIALLY
THE SAME AS BEFORE



- PREVIOUS STEPS REPEATED TO MANUFACTURE THREE 30m BEAMS
- 30 m BEAMS STORED TEMPORARILY

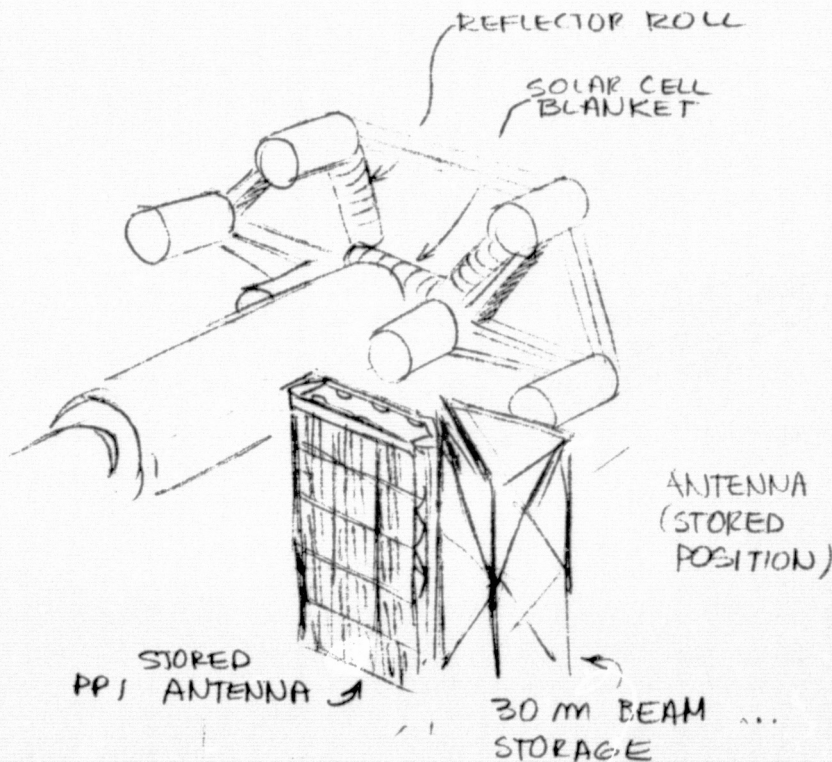
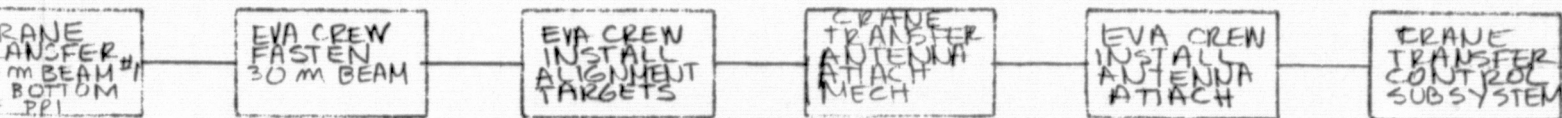


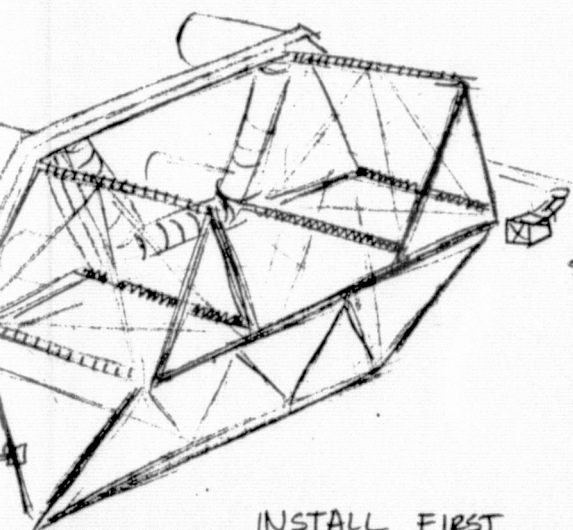
Figure 8-4. TA-2 (Sheet 9 of 10)

FOLDOUT FRAME \



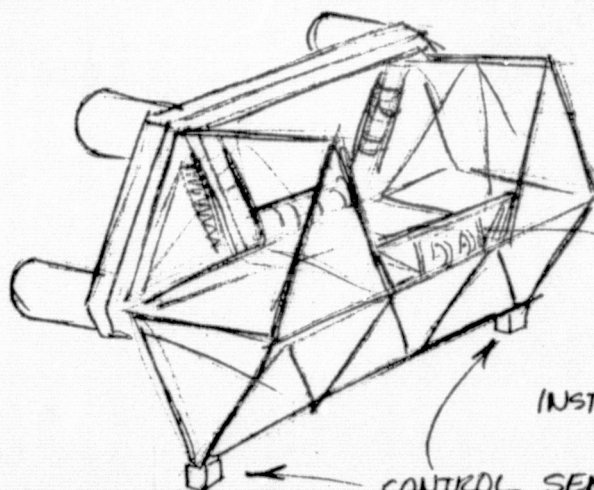
• FASTENING CONCEPT T.B.D

• AS THE PPT BEAMS ARE CONSTRUCTED, ALIGNMENT SIGHTINGS ARE MADE REGULARLY TO AVOID BEAM WARPAGE



EVA WORK STATION

INSTALL FIRST 30 m LENGTH BEAM



ANTENNA ATTACH

INSTALL EQUIPMENT

CONTROL SENSOR PACKAGES

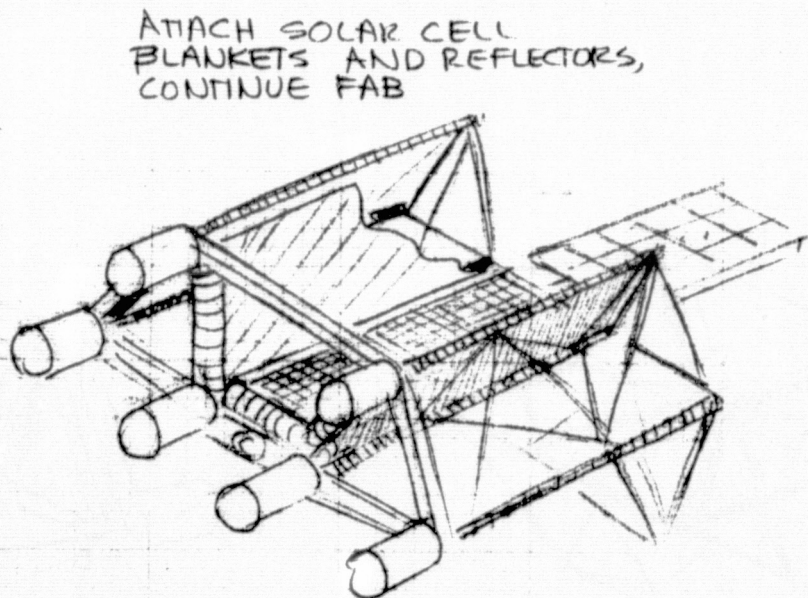
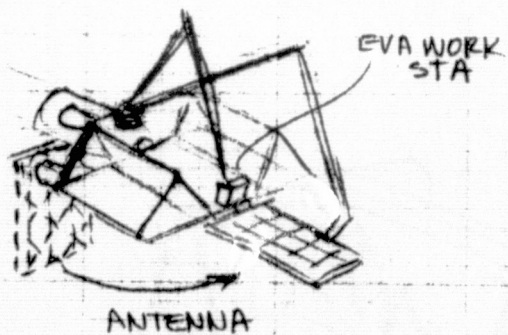
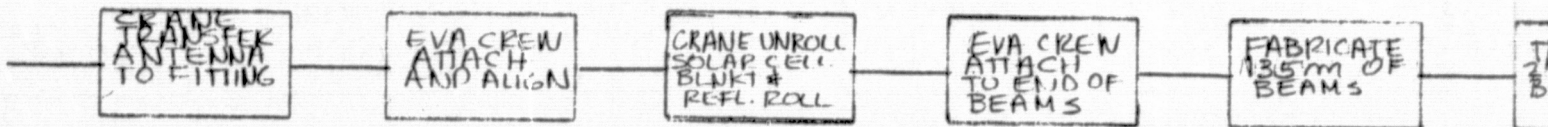
④

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FOLDOUT FRAME 2



FOLDOUT FRAME 3

TRANSFER
2ND 30m
BEAM

EVA CREW
ATTACH 30
m BEAM

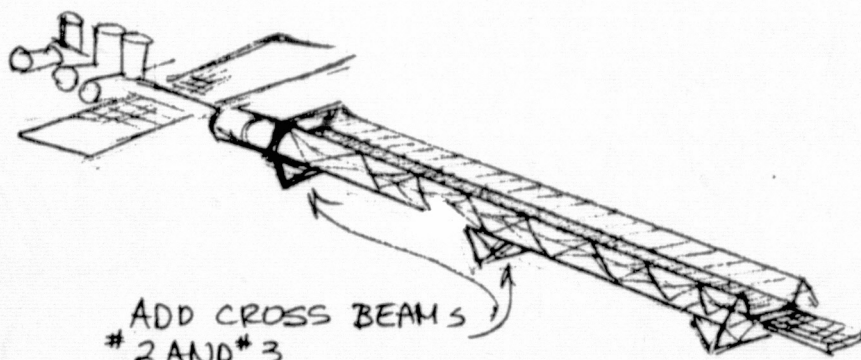
■ CONTROL
SENSORS
INSTALLED

FABRICATE
ADDITIONAL
125m

TRANSFER
3RD 30m
BEAM

EVA CREW
ATTACH
30m BEAM

→ CHECKOUT
& TEST



ADD CROSS BEAMS
#2 AND #3

FOLDOUT FRAME 4

⑤

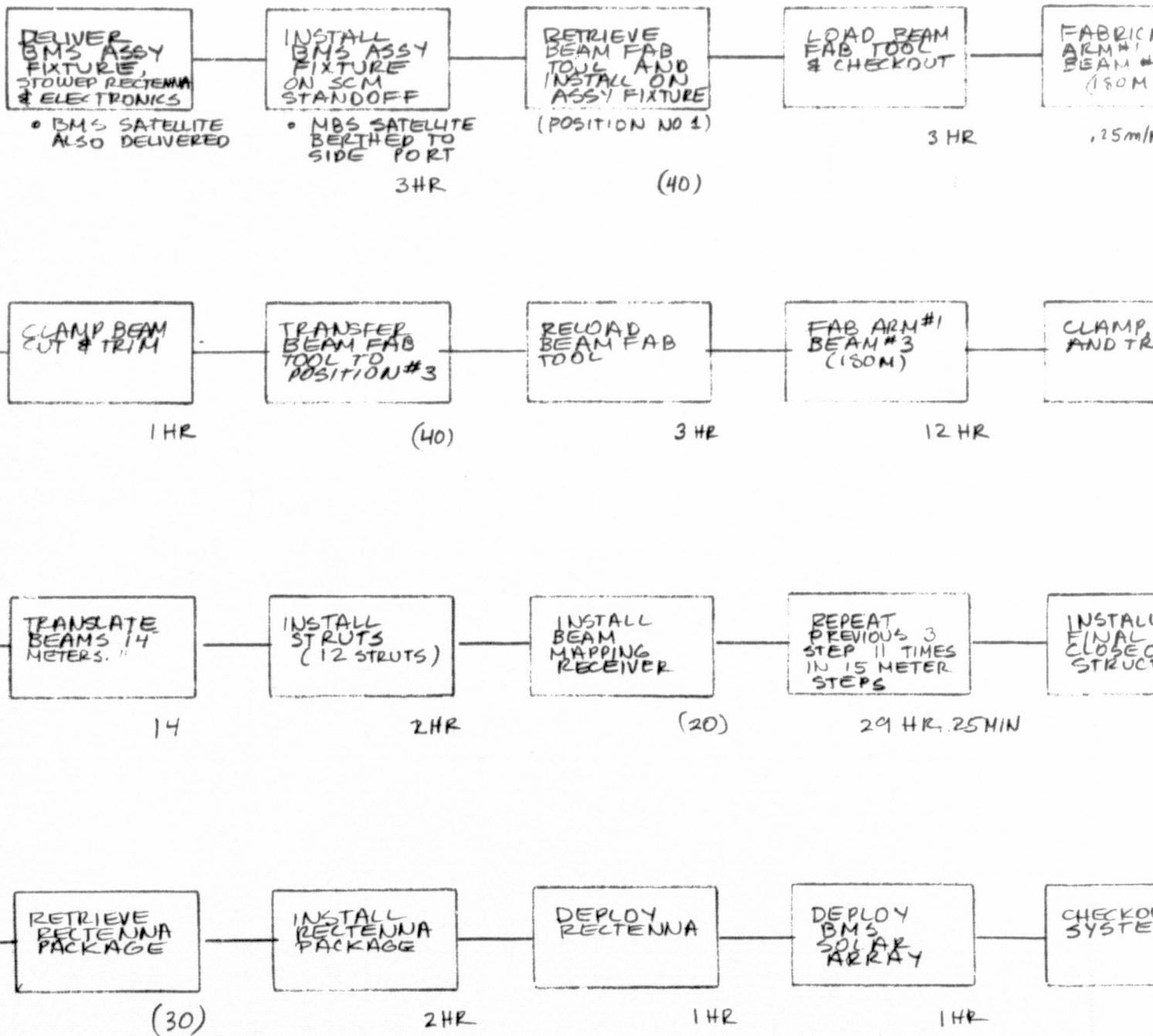
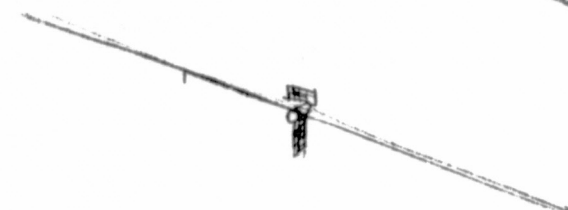
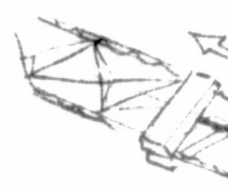
RE#

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12-8-76

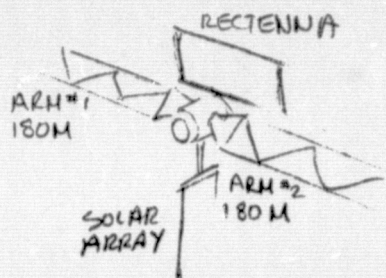
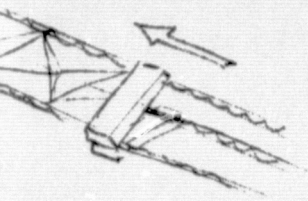
5.1

BEAM MAPPING SATELLITE (BMS) FABRICATION & ASSY HRS (MIN)

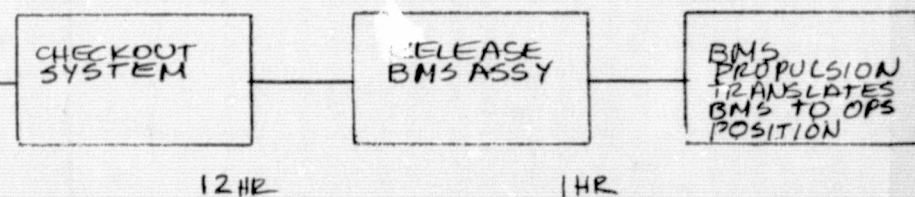
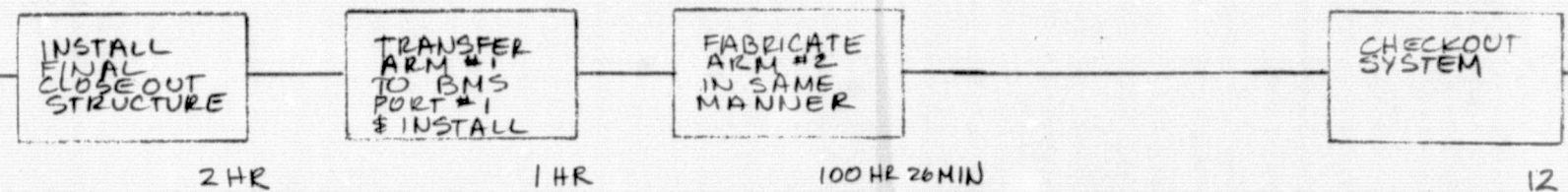
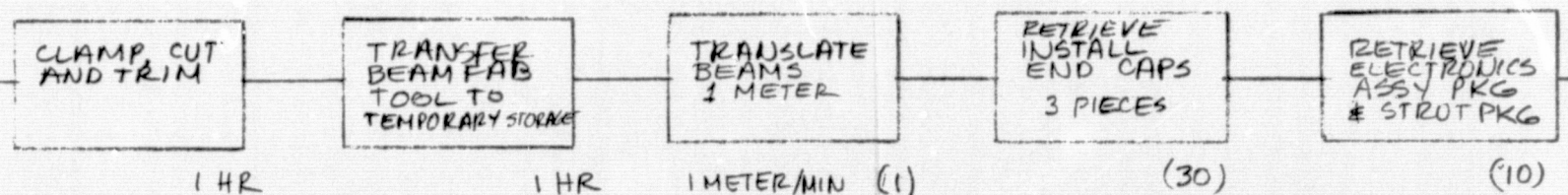
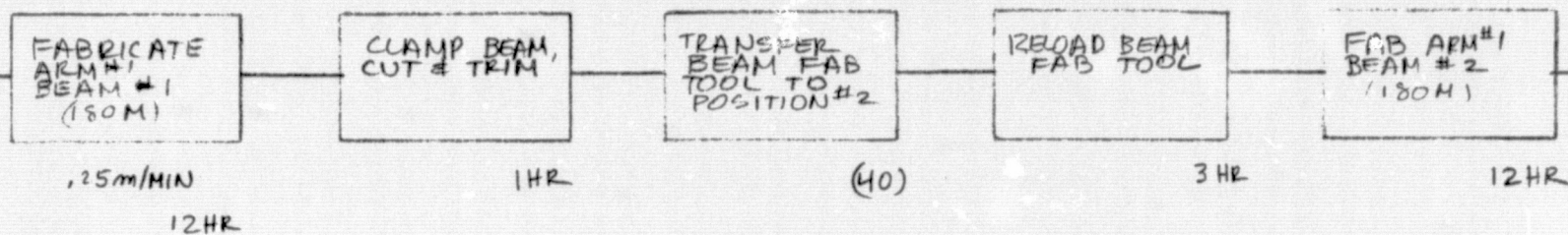


FOLDOUT FRAME \

Figure 8-5. Beam Mapping Satellite



130 STRUTS PER ARM
3 PILOT BEAM TRANSMITTERS
(1 CENTER, 1 EACH END)
25 BEAM MAPPING RECEIVERS



$$\Sigma = 233 \text{ HR } 22 \text{ MIN}$$

$$= 3.4 \text{ WEEKS}$$

FOLDOUT FRAME 2

REH 5-5-77

Section 9
38 kWe POWER MODULE

9.1 FUNCTIONS AND REQUIREMENTS

The function of the Power Module is to provide required quantities of electrical power, initially to Shuttle/Spacelab combinations and, subsequently, to the SCB in its early years prior to TA-2 antenna testing. Representative power requirements as a function of calendar year are presented in Figure 9-1. TA-1 antenna testing is intermittent, and is timed in conjunction with Spacelab experiments. A 25-percent contingency for potential load growth is applied to all loads except the TA-1 antenna tests, which are relatively well defined. In mid-1985, the Construction Shack comes up to complete the SCB assembly. The housekeeping power requirement is then 6.93 kWe without contingency, as follows:

<u>Item</u>	<u>Power Requirement (kWe)</u>
1. ECLSS	3.12
2. Data Management & Communications	1.20
3. Guidance and Control	0.81
4. Lighting	1.20
5. Miscellaneous	0.60
	6.93 kWe

The Sensor Lab and SPDF loads are intermittent in nature and are timed as shown. The Power Module size selected to meet these requirements is 38 kWe average to the load buses, rated at beginning-of-life (BOL).

The Power Module is required to free-fly for periods up to 90 days between Shuttle visits, prior to introduction of the Construction Shack. The Power Module is also required to provide peak loads of 75 kWe for TA-1 antenna testing for periods on the order of 15 minutes (in the sunlight), once each orbit; however, other loads can be reduced to about 20 kWe average (for the shuttle, spacelab and limited spacelab experiment maintenance) during antenna test periods.

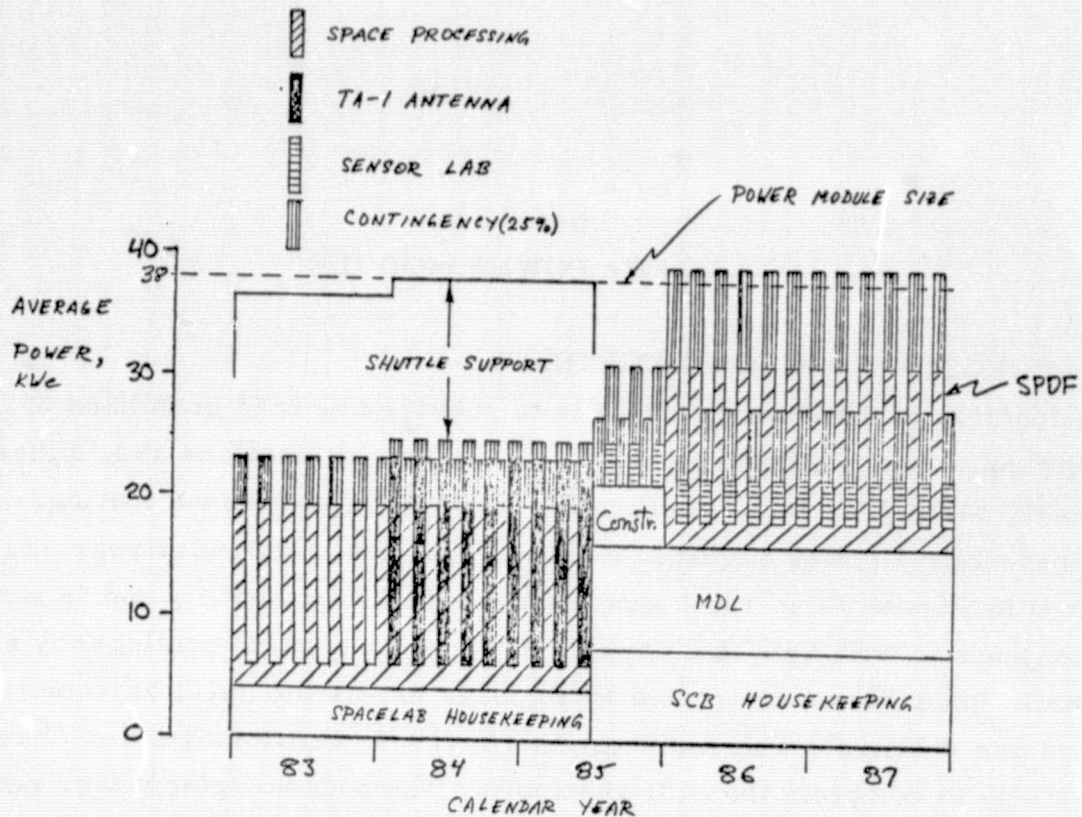


Figure 9-1. Power Module Requirements

9.2 CHARACTERISTICS AND OPERATION

The characteristics of the Power Module are presented in Table 9-1 and the weight in Table 9-2. The Power Module is illustrated in Figure 9-2, in conjunction with the Shuttle Orbiter and the Space Construction Module in a Shuttle-tended mode. The Power Module basically consists of: (1) a 15.5m-long cylindrical shell with a 5.5m-long telescoping cylinder at either end with the 2-axis gimbals at the solar array juncture; (2) two solar array wings, each consisting of four SEPS units, and (3), two deployable radiators.

A block diagram of the Power Module electrical power system is shown in Figure 9-3. The 26 VDC accommodates Shuttle and other existing equipment, the 112 VDC feeds the inverters and new equipment (e.g., mission hardware, as appropriate) and the 76 VDC is for TA-1 and other microwave testing. The nominal battery depth of discharge is 16.2 percent. The 1004 m² solar

(Continued on Page 9-8)

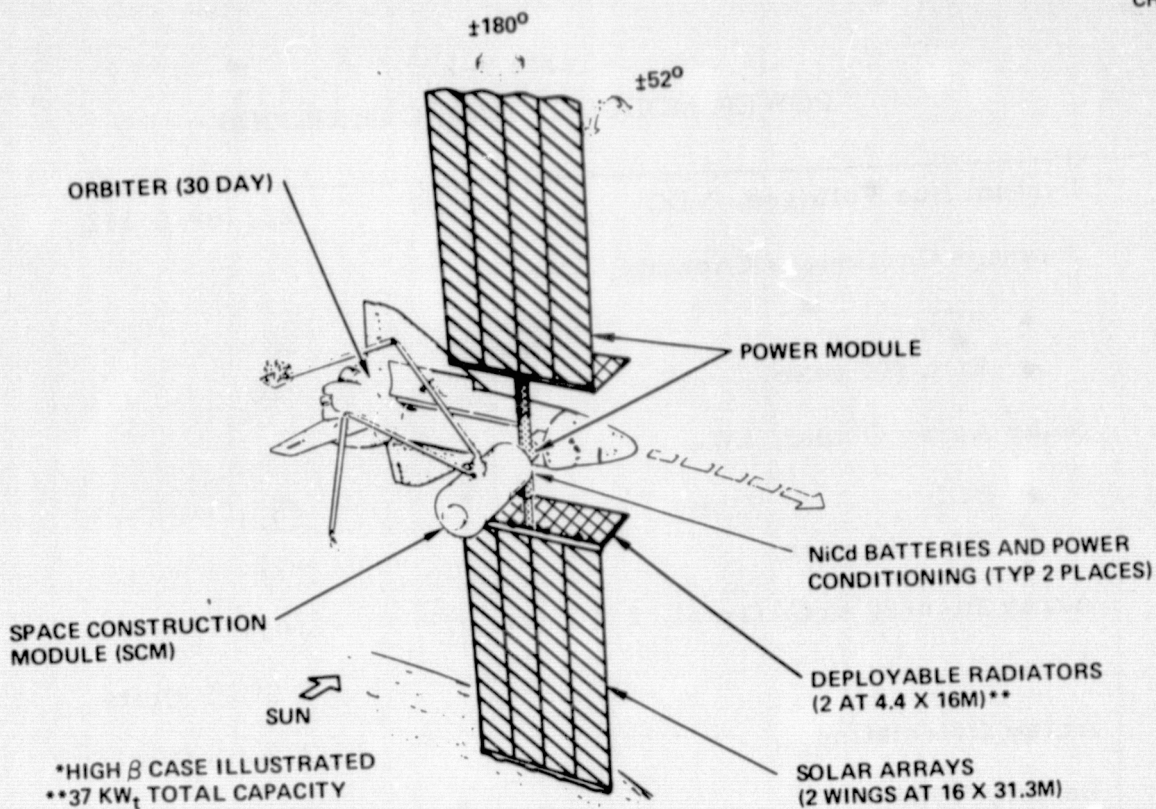


Figure 9-2. Power Module (38 kW e Average)*

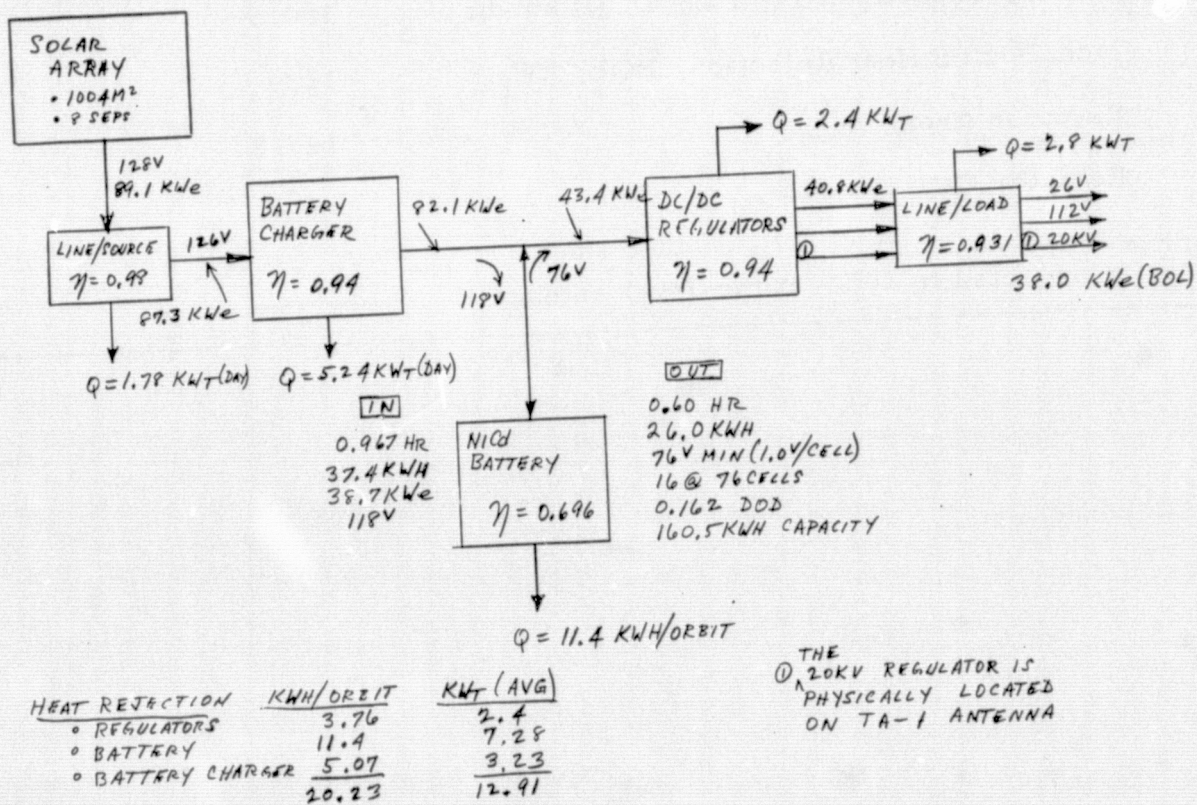


Figure 9-3. Power Module Block Diagram

Table 9-1
POWER MODULE CHARACTERISTICS

System Bus Voltages, VDC	26, 76* & 112
Average Continuous Capacity, kW _e	
• BOL	38
• EOL (10 year)	~30
Solar Array Output, kW _e	
• BOL	89.1
• EOL (10 year)	~71
Array Blanket Area (Total, 2 Wings), m ²	1004
Array Concept	8 SEPS Units
Array Orientation	2-Axis Gimbal
Battery Type	Ni Cd (110A-H Cells)
Battery DOD, Nominal, %	16.23
Battery Capacity (16 @ 100% DOD) kW-hr	160.5
Module EPS Heat Rejection, BOL, kW _t	12.9
Radiator Area, m ²	140.8
Radiator Capacity, kW _t	37.0
*Converted to 20KV at the TA-1 antenna	

Table 9-2
POWER MODULE WEIGHT

Item	Weight, kg
Solar Arrays (SEPS Wings, 8 @ 192)	1,540
Battery (16 @ 370.5)	5,928
Battery Chargers	104
Load Regulators	177
Wire	400
Support Structure and Gimbals	1,701
Radiator/Thermal Control	957
Attitude Control and Orbit Keeping	429
Avionics	59
Total	11,295 kg

The Power Module contains a limited communication and navigation system, and an RCS, which permits it to free-fly between Orbiter visits. A 4.4-meter-wide 16-meter-long radiator panel is mounted along each of the two array support beams. The radiator panels are hinged as shown in Figure 9-5 so that they fit within the cargo bay clearance envelope. Mounted as they are, outboard of the gimbals and normal to the plane of the arrays as shown in Figure 9-6, the radiator panel surfaces are always parallel to the sun line and their heat-rejection capacity is maximized.

9.3 STRUCTURAL/MECHANICAL CONFIGURATION

The proposed Power Module concept uses eight baseline SEPS arrays shown in Figure 9-4. Four arrays are mounted on each of two 16-meter-long support beams. The launch packaging arrangement is shown in Figure 9-5. Support beams are attached to fittings on the exterior surface of a 15.5-meter (51 ft)-long cylindrical shell which contains two berthing ports at its center. Telescoped inside each end of the 15.5-meter cylinder is a 5.5-meter-long cylinder which contains a two-axis gimbal at its outboard end.

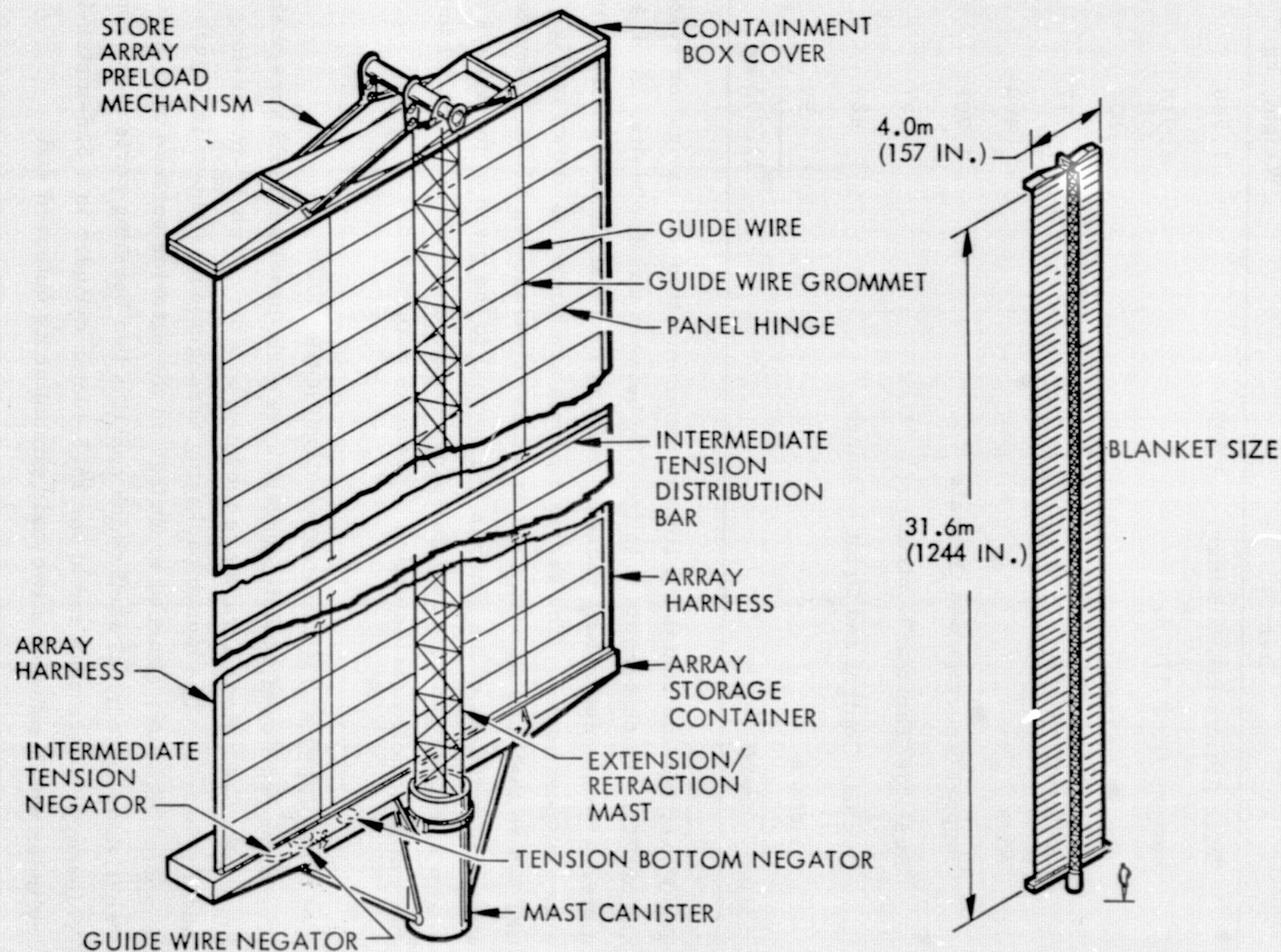


Figure 9-4. SEPS Solar Array Wing Configuration

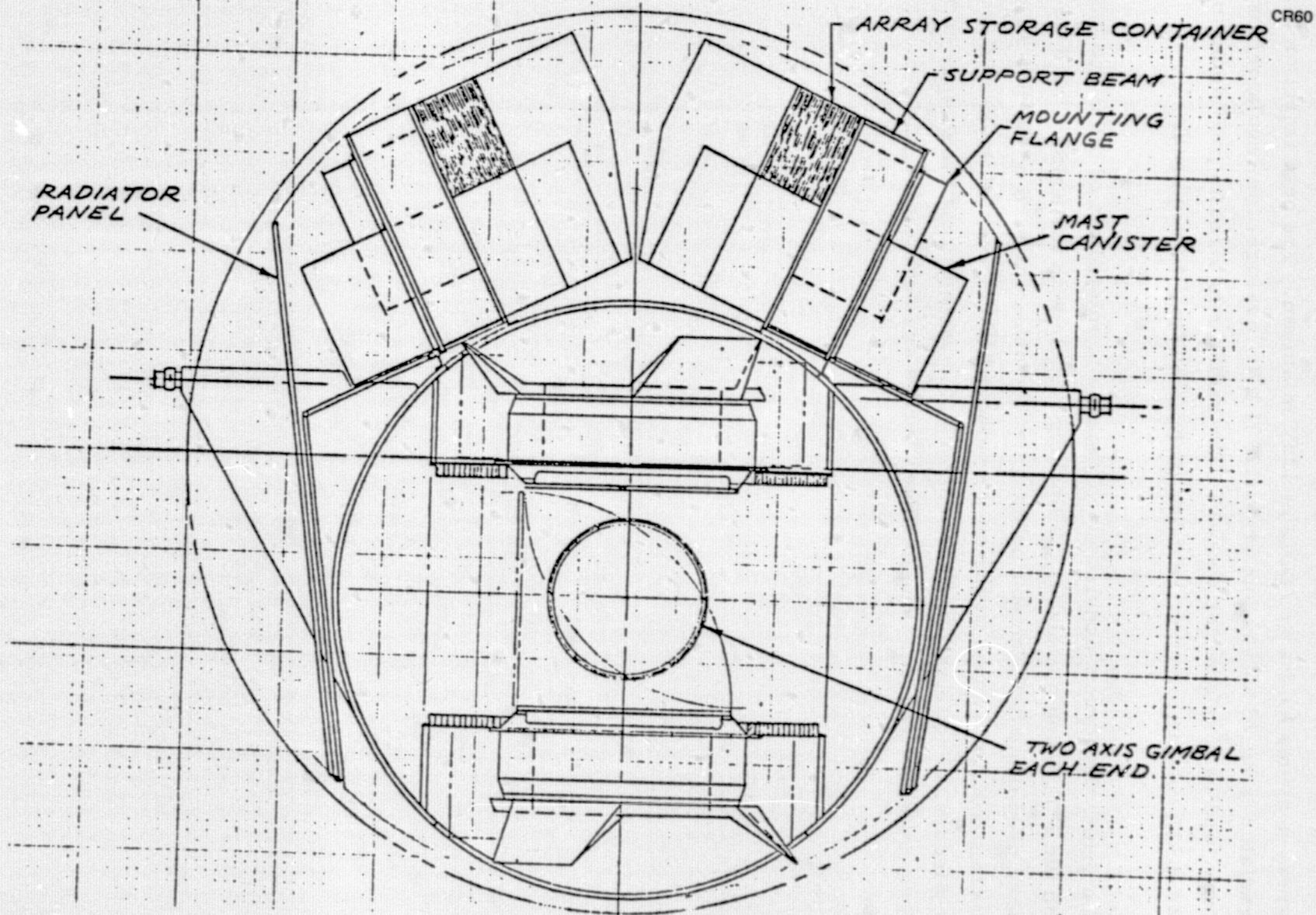


Figure 9-5. Power Module Launch Packaging

(Continued from Page 9-2)

array provides 89.1 kWe (BOL) at 128V minimum. The heat rejection to the radiator is 20.23 kW-hr per orbit (12.91 kWt, average) for regulator, battery, and battery charger cooling. The radiator is sized to provide up to 24.1 kW cooling for other vehicles (e.g., Shuttle), as required.

On orbit, the Power Module launch package is rotated out of the cargo bay on the PIDAS and berthed at the Docking Module with the RMS. The RMS is then used to mate the mounting flange at the center of each of the two support beams with the gimbal fitting on the outboard end of each of the two telescoped cylinders. The telescoped sections are then extended and the flanges on the inboard ends secured. The deployed arrays are shown in Figure 9-6.

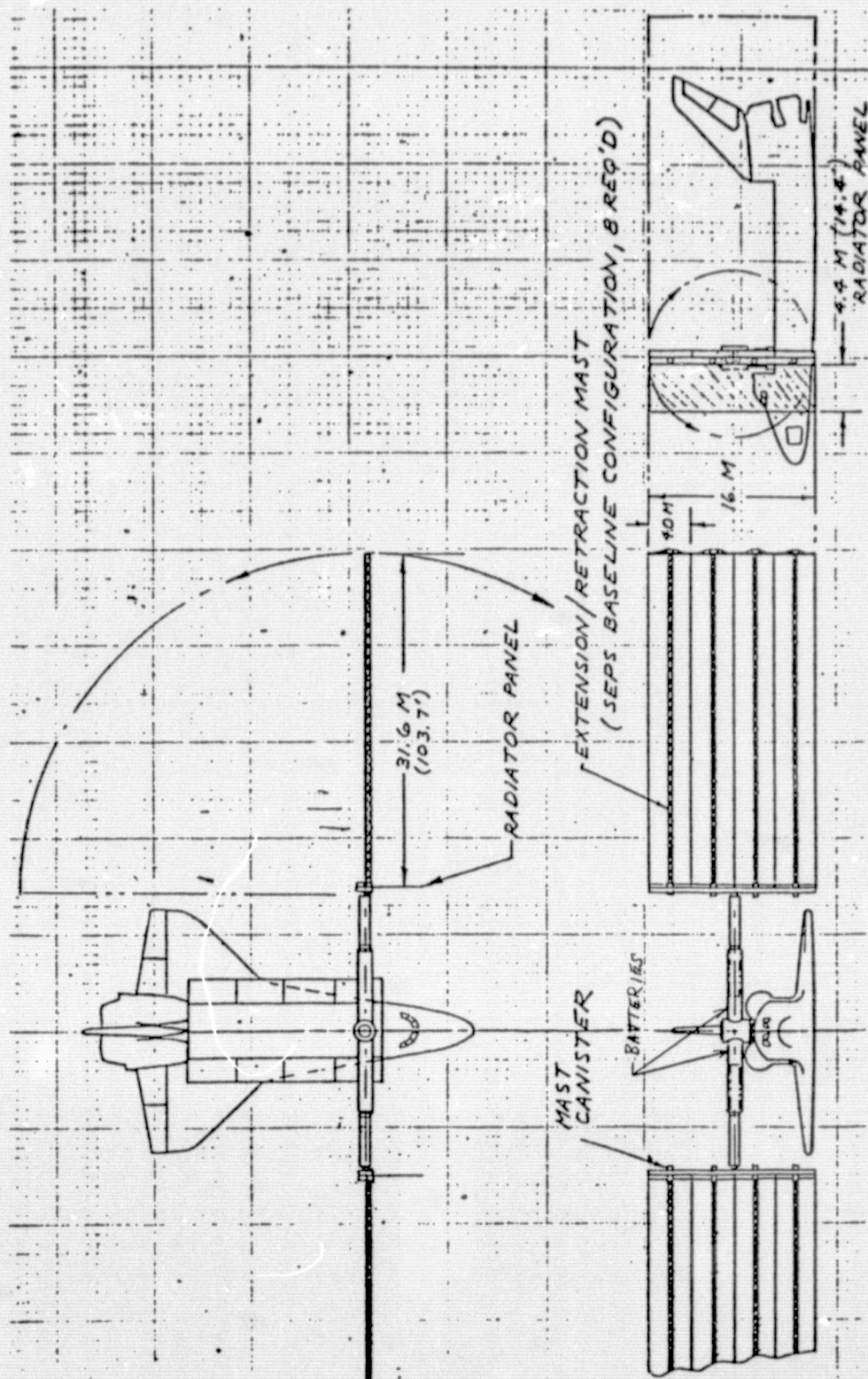


Figure 9-6. Power Module Configuration

Section 10
ENVIRONMENTAL CONTROL AND LIFE SUPPORT
SUBSYSTEM CONCEPTS MODE STUDIES

10.1 INTRODUCTION AND SUMMARY

This section presents a description of key trade studies performed to select ECLSS concepts for the low cost SCB. The selected ECLSS was then defined in sufficient detail to allow a valid assessment of its impact on the various candidate SCB program options. Key guidelines and trade factors used in the trade study are also included.

Basis for the ECLSS tradeoffs was cost and therefore all major impacts of implementing a concept was reduced to a cost value. Costs given are equivalent costs and do not reflect only ECLSS costs. They also include such items as Shuttle launch costs, ground support contractor costs, and costs of power and structure support systems.

Not all possible concepts for each ECLSS function were evaluated, only key trades were performed. These included the following.

- A. Airlock pumpdown versus expendable atmosphere (i. e., or inboard dump)
- B. Open oxygen loop CO₂ control concept (LIOH versus solid amine)
- C. Oxygen recovery tradeoff
- D. Water recovery tradeoff

Results of these trades are summarized below.

The airlock pumpdown trade essentially was a tradeoff of resupply penalty of the expendable concept versus crewtime cost for pumpdown. Crewtime was charged against pumpdown for the time spent by the EVA crew waiting for the airlock pumpdown (12 minutes/EVA). Also the pumpdown system had a lower initial hardware cost because of the large number of resupply tanks for O₂ and N₂ required for the make up of the atmosphere dumped overboard for each EVA. On the basis of these advantages, the pumpdown concept was selected for SCB.

The open-loop CO₂ control trade favored the solid amine (HS-C) concept primarily due to lower initial cost as this concept also controls humidity and, therefore, the humidity control unit can be eliminated. Resupply weight of solid amine is slightly higher than LIOH. This resupply is for the water which the solid amine dumps overboard during the humidity control process. However, the lower initial cost offsets the higher resupply weight and the solid amine concept is selected.

The closed oxygen concept has a higher initial cost than open oxygen, however, open oxygen has a very high resupply requirement. Up to \$41 million can be saved initially with open oxygen. However, at medium launch costs \$1,630/kg (\$741/lb), about \$21 million per year resupply costs will be incurred by the open system. This results in closed oxygen trading favorably after about two years. Based on the much lower initial cost for open oxygen, it is selected for the initial low-cost SCB. However, it is recommended that the closed oxygen concept be retrofitted after initial operations because of the large cost savings which can be realized in the continuously manned multiyear program.

10.2 TRADEOFF FACTORS AND GUIDELINES

The trade factors and guidelines form the basis for the trades performed in the ECLSS area. Cost was used as a normalization factor to allow comparison of the many effects of implementing candidate approaches.

10.2.1 Cost Trade Factors

Cost is a very pertinent basis for comparison because minimizing cost is a primary program guideline. This approach is conveniently applied in the trades because implementation of a candidate ECLSS approach causes perturbations in the system which can be converted to cost. Examples are program cost changes due to increasing module length for volume increases, increasing solar array size for power increases, and increasing thermal control assembly size for increased cooling. All of these incremental changes in supporting resources are reflected in cost changes. Below, the cost factors used in the tradeoffs will be presented along with the rationale for their values.

10.2.1.1 Crew Time

Crewtime is a valuable resource in the SCB program and a concept using a significant amount is charged for that resource. Basis for crew time cost is ground support contractor cost estimates of \$35m/year to support a seven-man crew. Considering the available crew work time, a factor of \$4,674/manhour results.

10.2.1.2 Module Volume

Several factors were derived for launch volume which depended on:

(1) stretching modules or providing new modules and (2) type of basic SCB module. Three separate factors were derived: initial launch costs used the basic SCB module costs, resupply costs used the logistic costs, module stretching costs were based on cost differentials for increasing the length of outer shell, meteoroid shield, and floor structure. Stretching the module does not affect the cost of such items as end cones and docking ports.

Cost factors for an additional module simply divides the cost of a module by the available volume. Table 10-1 shows factors derived for volume penalties.

Table 10-1
MODULE VOLUME COST FACTORS

Type Module	Cost Factors, \$/m ³ (\$/ft ³)			
	Stretch Module		Additional Module	
Basic SCB module (initial launch)	51,040	(1450)	246,400	(7000)
Logistics	24,040	(685)	205,320	(5835)

10.2.1.3 Power

An increase in power system size may be necessary when a subsystem concept is chosen which requires more power and power timelining is not possible. This increase in size is accompanied by a delta cost which should be charged to the subsystem concept. A cost factor was derived which includes deltas in DDT&E costs and production costs of the solar array and

associated equipment. The resulting cost factor is \$8,770/watt-five years. The cost is determined as a function of time since the power would be available for another use when not used by the candidate subsystem concept.

10.2.1.4 Hardware Costs

Equipment costs were derived from two sources; (1) "Advanced Spacecraft Subsystem Cost Analysis, MDC E0027," and (2) RLSE data prepared under Contract NAS9-14782 and called "Thermal Control and Life Support Subsystems Parametric Ddata for Space Station" (see References 2 and 3). The first listed reference contains cost estimating data primarily from projects Mercury and Gemini. The second listed, Reference 2, emphasizes closed-loop concepts, especially RLSE concepts and is an up-to-date source. The second source was used when it became available and was used primarily for the oxygen and water recovery trades. All incurred costs were used including subcontractor and prime contractor costs.

10.2.1.5 Launch Weight Penalties

Costs for launching equipment and expendables was treated parametrically because related costs per launch are not completely defined at this time. Therefore, the effects of launch costs were used only to obtain a sensitivity to that parameter.

10.2.2 Guidelines

General guidelines were used to establish overall requirements and philosophy of the SCB subsystem designs. Additional assumptions were made for the basis of the trades and interface data was obtained from related SCB studies in other subsystem areas. These guidelines and assumptions are listed below.

- A. Low initial cost is of primary concern - this comes from "SCB Design Guidelines and Criteria, JSC-11867," Reference 1.
- B. Regenerative Life Support Evaluation (RLSE) concepts were used for regenerative ECLSS concepts, because these are currently the concepts under development and their use can save substantial DDT&E costs.

- C. Subcontractor and Prime contractor costs shall be included. This is necessary to assess the full cost impacts of implementing an ECLSS concept.
- D. 5633 kg/m^2 (8 psia) cabin pressure was assumed. This assumption is consistent when large amounts of EVA is planned. Major drawbacks include Orbiter interface at 1 atmosphere and special support requirements for life science experiments.
- E. Extensive EVA was assumed, amounting to two, two-man EVA shifts occurring six days per week. This is a result of EVA task analyses in the study.
- F. Two tank sets were assumed for expendable fluids.
- G. LIOH and solid amine were assumed as prime candidates for open oxygen loop CO_2 control. LIOH is the current choice on the baseline Orbiter, solid amine (HS-C) is expected to be used on the 30-day Orbiter version.
- H. 180-day maximum resupply period - this guideline comes from Reference 1.
- I. Seven-man crew - this is a result of SCB studies on crew activity needs.

10.3 TRADEOFF ANALYSES

This section describes the detailed approach and results for each of the trades performed. The trades are reported on in the logical sequence to allow results of one trade feeding into the next; for example, the open oxygen CO_2 control concept trade selection is traded against the closed O_2 concept.

10.3.1 EVA Airlock Pumpdown

Current guidelines for the SCB program calls for extensive EVA, making the expendable approach costly from a resupply standpoint. This trade was performed to compare a pumpdown concept versus an expendable overboard dump concept. The pumpdown approach includes the development and unit costs for the pump and related hardware. The expendable concept is simpler and less initial cost is expected when compared to pumpdown. However, greater resupply is necessary.

The pumpdown system concept initially pumps airlock air into the cabin down to an airlock pressure of 352 kg/m^2 (0.5 psia). Prior to the crew opening the outer airlock hatch the remainder of the air is expended overboard with an equalization valve. Upon return from EVA, the crew will secure the outside hatch and repressurize the airlock by equalizing airlock and cabin pressures with a second equalization valve. The small amount of atmosphere which is dumped overboard is made up by the normal cabin atmosphere supply system.

The expendable airlock concept dumps the entire airlock atmosphere overboard with each operation. Repressurization is accomplished by adding O_2 and N_2 directly from the supply system. This approach is preferred over cabin pressure equalization to allow airlock pressurization with O_2 in the event EVA shift breaks become necessary, thereby reducing N_2 loss.

Overpressure relief valves are provided in both airlock concepts to protect the structure from events such as a suit pressurization subsystem failure which could allow O_2 to escape and overpressurize the airlock.

Table 10-2 presents detailed trade assumptions not covered in Section 10.2.

Table 10-2
DETAILED ASSUMPTIONS – AIRLOCK PUMPDOWN TRADE

Airlock volume	5.68m^3 (200 cu ft)
Module volume	177m^3 (6240 cu ft)
Diameter resupply tanks	0.91m (3 ft)
Pump efficiency	20%

A suboptimization was performed to establish the pumpdown time. This is essentially a trade between pump peak power and crew time. Two methods were used to account for power; (1) average power assuming a 100 percent efficient power scheduling and (2) peak power assuming power system for pumping cannot be used for any other needs. Figure 10-1 shows the results of this trade for the average power case. Results show that crew time is the driving factor and a pumpdown time equal to crew EVA preparation time is indicated. Figure 10-2 gives the same trade but with power cost based on

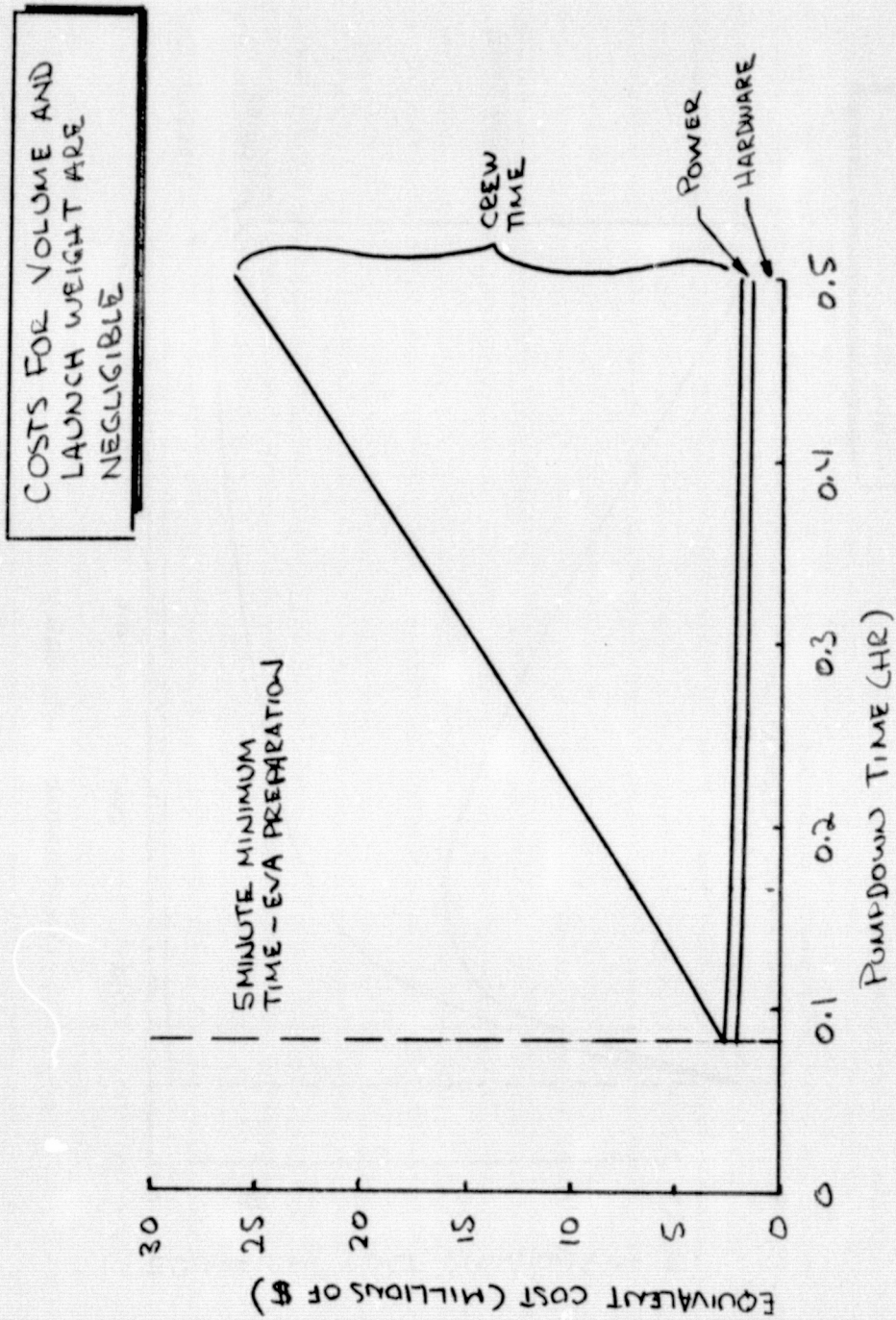


Figure 10-1. Airlock Pumpdown Time Optimization Power Cost Based on Daily Average

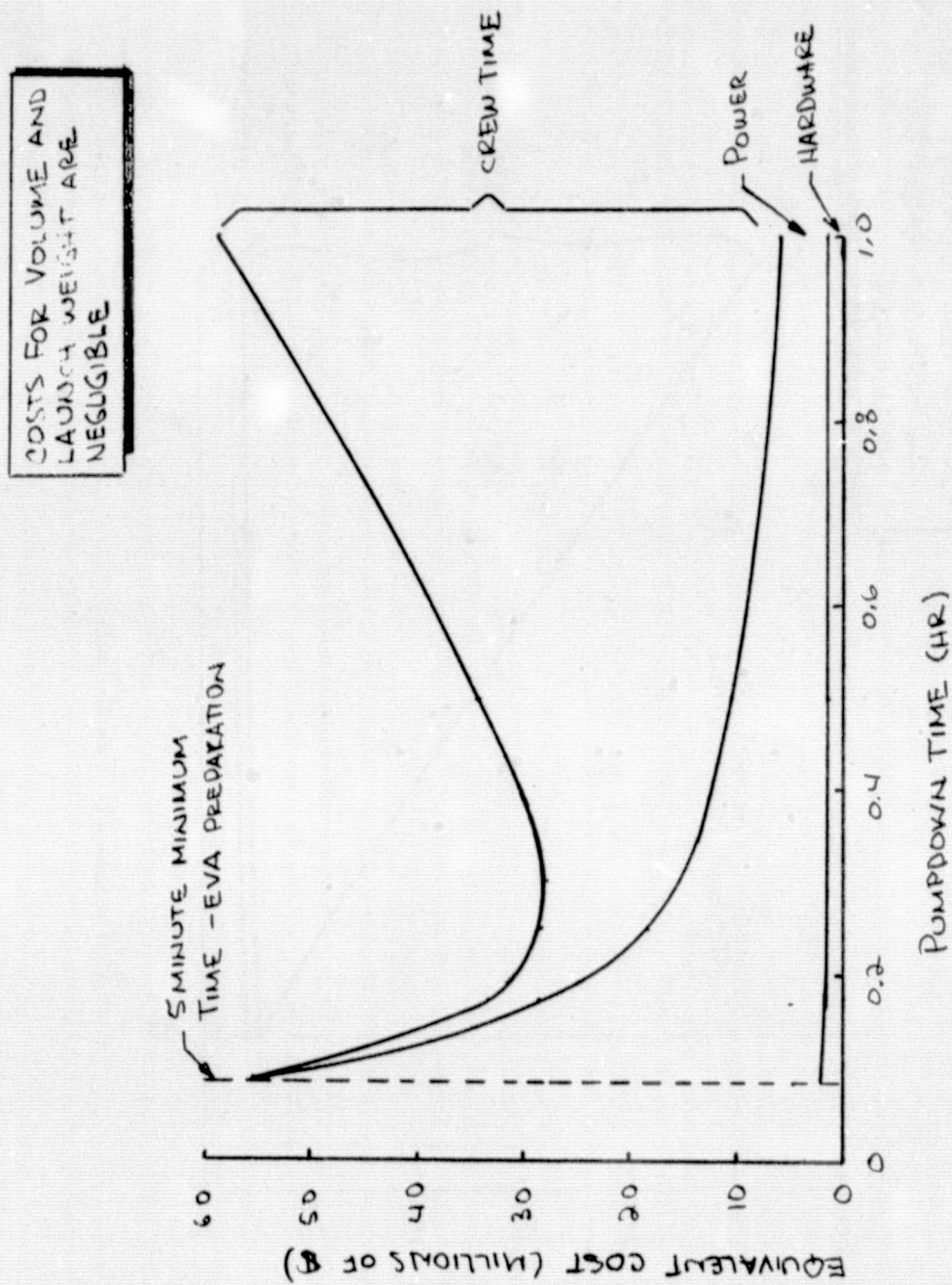


Figure 10-2. Airlock Pumpdown Time Optimization Power Cost Based on Dedicated Peak Demand

the peak power demand. This trade shows about 18 minutes as the most cost effective.

In actual practice, it is expected that the optimum time will lie somewhere between the extremes of 0 to 100 percent efficient power scheduling. Therefore, an intermediate value of 12 minutes was selected.

Pumpdown trade results are shown in Figure 10-3 for zero launch costs, high volume penalty charged initially, and average pump power costing. Refer to Section 2 for rationale of these costing assumptions. The purpose of this figure is to show the relative costs both initially and after five years. Volume costs are due mainly to high-pressure resupply tanks. Crew time penalty is time charged against pumpdown when the crew is waiting in the airlock for the pumping operation to be completed, 7 minutes per operation. Hardware costs for the expendable system are higher than for pumpdown because of the large number of tanks required, 46 required for two logistics modules. Four tanks are required for the pumpdown system. The pumpdown trade shown in the Figure 10-3 shows that the pumpdown system has a much lower initial cost and still costs about half of the expendable system after five years.

Figure 10-4 shows the same trade but with launch costs added and for both high and low volume penalties. Total cost for a two-year time period is shown. Results show that the pumpdown system is more cost effective when launch costs are added. The only combination of cost assumptions which would make the expendable concept more cost effective would be (1) low volume penalty, (2) zero or very low launch costs and (3) high power penalties. As was shown in the previous figure, the cost of average power is small compared to the other costs. Only very inefficient power scheduling would make power costs significant.

Based on the above trade data, the pumpdown system is selected because it saves \$2 to \$22 million initially and will become more cost effective for greater use or if launch costs are included in the trade.

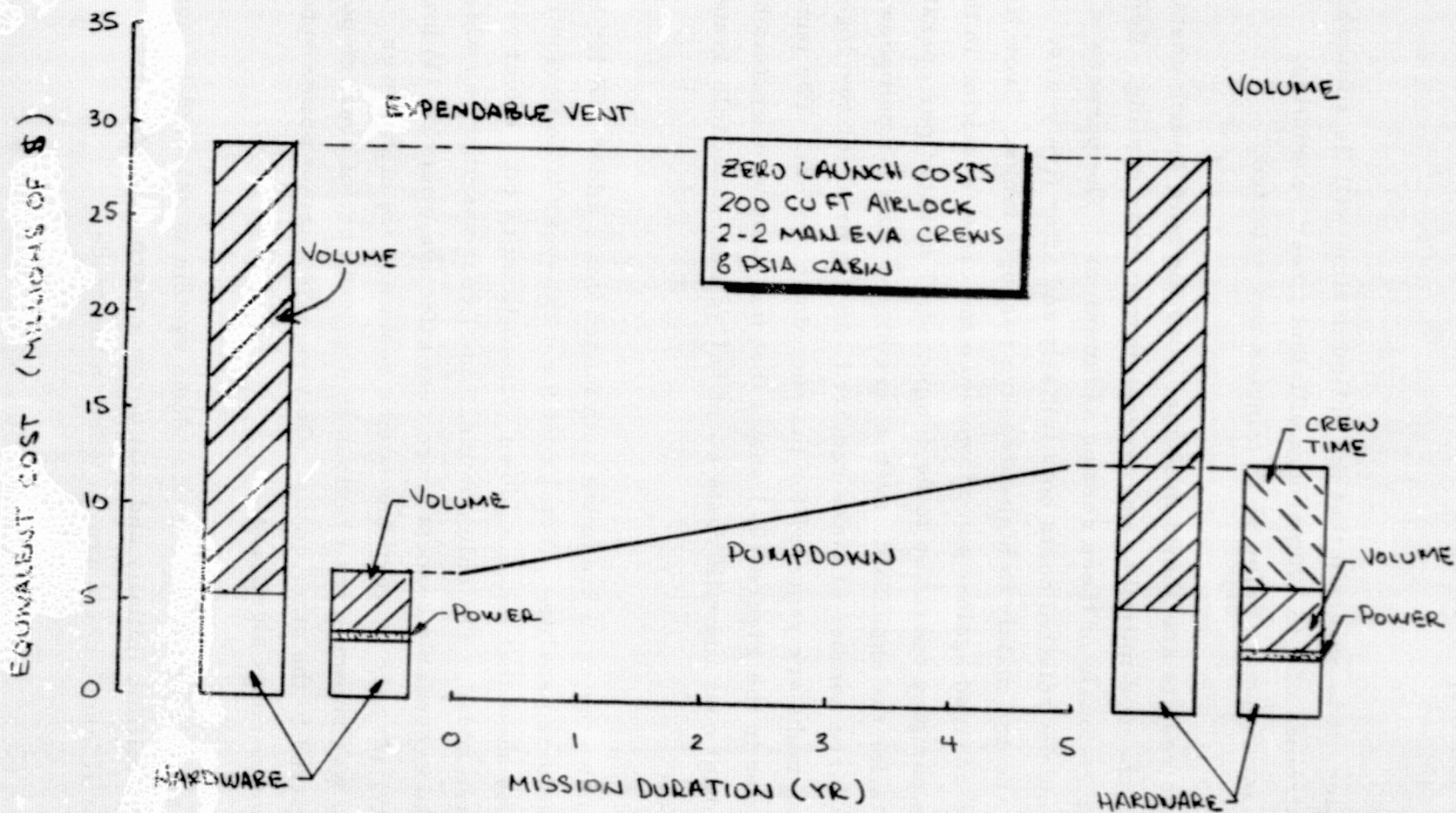


Figure 10-3. Expendable Versus Pumpdown Airlock Concept High-Volume Penalty — Charged Initially

200 CUFT AIRLOCK
2-2 MAN EVA CREWS
8 PSIA CABIN
2 YEAR MISSION

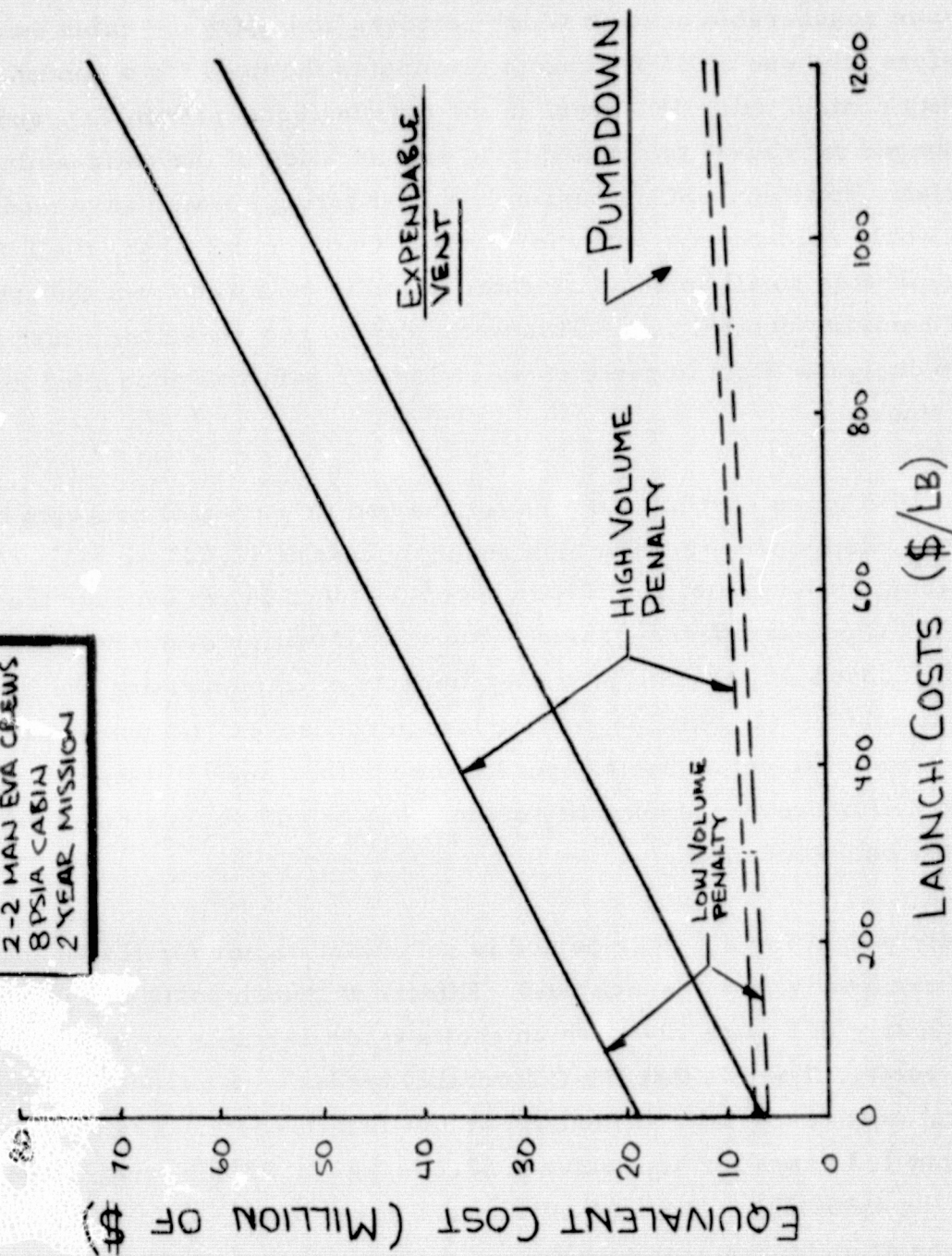


Figure 10-4. Expendable Versus Pumpdown Airlock Concept Launch Cost Effects

10.3.2 Open Oxygen Loop CO₂ Control Concept (LIOH Versus Solid Amine)

This trade considers two major candidates for CO₂ control in an ECLSS design not employing oxygen recovery. The current Orbiter chemical absorbent (LIOH) is compared with a regenerable solid amine (HS-C) concept which is being developed for the 30-day Orbiter. The solid amine concept is a vacuum regenerable concept which removes both CO₂ and cabin humidity. Therefore, the use of HS-C concept eliminates the need for a condensing humidity control unit. However, in the existing concept both CO₂ and H₂O are dumped overboard to space during regeneration of the solid-amine bed material. However, JSC is currently investigating a water save modification which would recover this water for reuse. On the other hand, the LIOH concept generates small amounts of water which is removed from the cabin air by the humidity control unit. This water can be recovered for reuse by most ECLSS designs. Cost impacts of water loss or gain was accounted for in the trade study.

Figure 10-5 gives cost breakdowns for the two CO₂ control concepts based on a low volume penalty and a high resupply cost of \$2,640/kg (\$1200/lb). Comparing initial costs, the HS-C concept is much lower because the cost of the HS-C unit is offset by savings due to elimination of the humidity control (condenser) assembly. Other impacts of implementing the HS-C concept include: (1) sensible cooling lost in condenser, (2) power savings due to more efficient higher temperature radiator, and (3) water resupply necessary for overboard humidity dump. Power and volume costs are about equal for both concepts.

Resupply weight for a 2-year period is somewhat higher for HS-C, but this cost corresponds to the launch cost. Effects of launch costs can be seen more clearly in Figure 10-6, which shows tradeoff points for a wide range of launch costs. It shows that HS-C favorably trades with respect to initial cost and total cost, especially at medium or low launch costs. Tradeoff points are about 1.3 years for high values, \$2,640/kg (\$1,200/lb) and 2.5 years for \$1,320/kg (\$600/lb). About \$4 million are saved for HS-C for all mission durations at no launch cost penalty.

7 MAN UNITS
WATER LOSS CONSIDERED
HS-C CONTROLS HUMIDITY
\$200/LB RESUPPLY
LOW VOLUME PENALTY

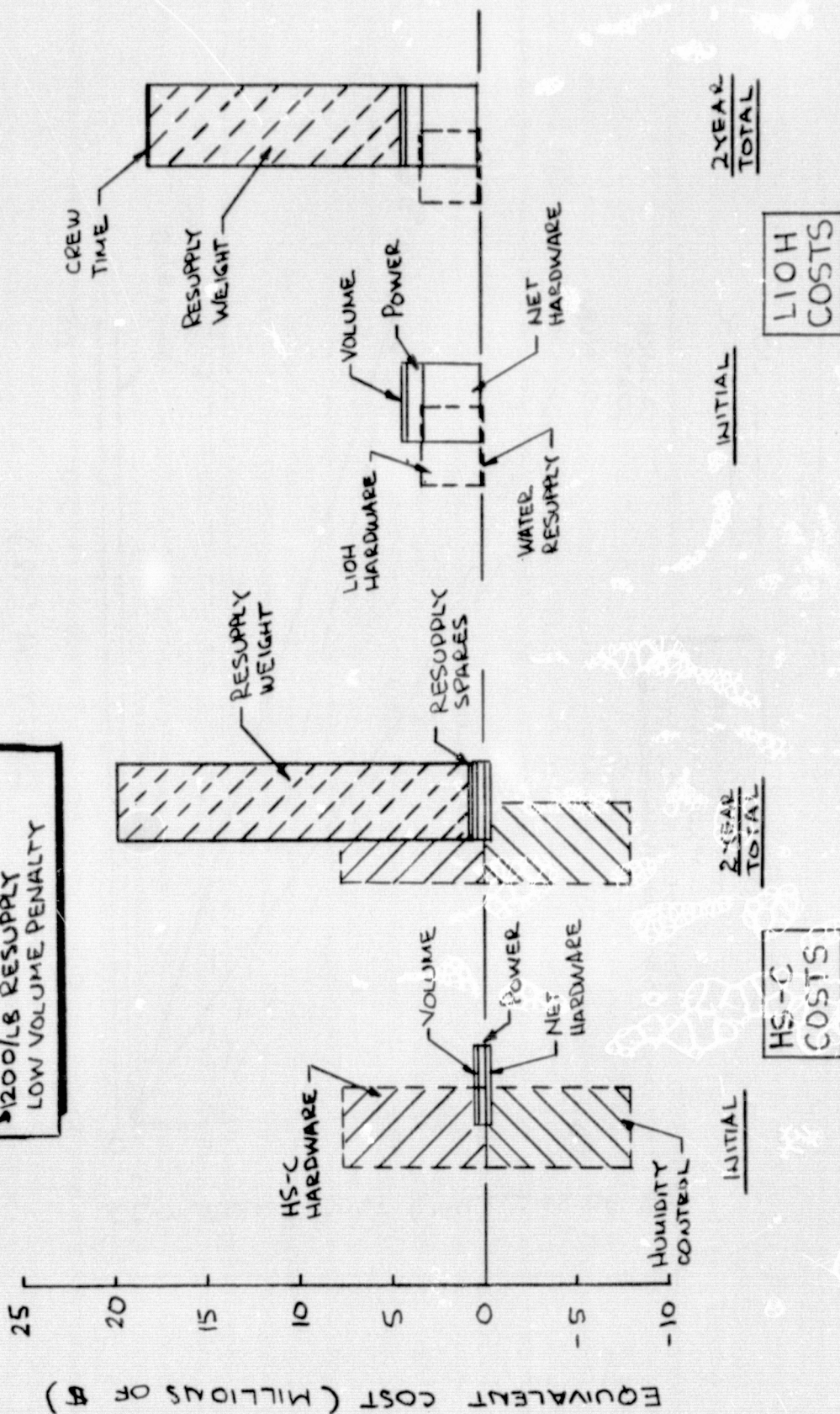


Figure 10-5. C27 Control Concept Trade Expendable (LIQH) Versus Regenerable (HS-C)

10-14

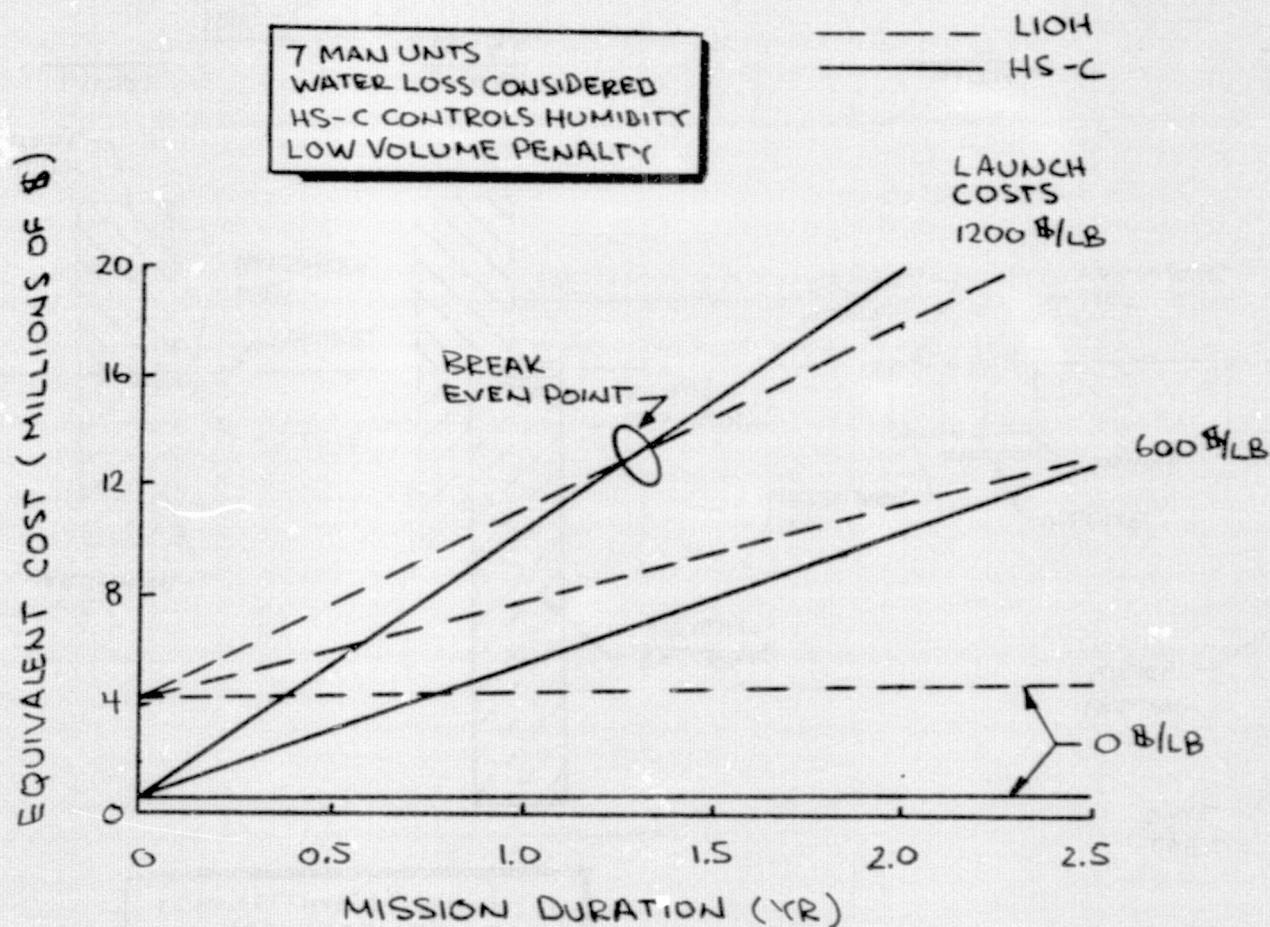


Figure 10-6. CO₂ Control Concept Trade Expendable (LIOH) Versus Regenerable (HS-C)

Based on lower initial cost and low total program cost early in the program, the HS-C concept is selected for SCB.

10.3.3 Oxygen Recovery Tradeoff

This trade compares the open versus closed oxygen recovery loop. Table 10-3 lists the assumed concepts; Figures 10-7 and 10-8 give block diagrams and mass balances for each concept, assuming closed water loop.

Table 10-3
OXYGEN RECOVERY TRADEOFF ASSUMED CONCEPTS

Function	Open Loop	Closed Loop
CO ₂ removal	HS-C solid amine	Electrochemical depolarized concentrator
Oxygen supply	High pressure gaseous	Water electrolysis
CO ₂ reduction	None	Sabatier reactor

The assumed design for open loop is similar to the expected 30-day Orbiter, the closed loop design is identical to the Regenerative Life Support Evaluation (RLSE) concept currently being developed by NASA/JSC. Data for performing the tradeoffs were obtained primarily from the Shuttle and RLSE programs.

Mass balances were performed for both candidate systems to identify the required performance of the systems and to account for impacts on resupply and penalties to other ECLSS assemblies. The mass balances, shown in Figures 10-7 and 10-8 take into account the impact of the planned EVA consisting of two, two-man EVA's lasting for 6 hours and occurring 6 days per week. It is assumed that the oxygen supply system provides EVA oxygen and the EVA condensate is processed by the ECLSS. Modifications were also made in the crew metabolic balance to account for EVA.

Figure 10-9 gives trade result costs initially and after 2 years for a low volume penalty and a launch cost of \$1,630/kg (\$741/lb). Initial costs for the closed oxygen concept are high because of hardware and power costs. However, at two years, the open concept has about equal cost because of the resupply costs for open oxygen. Initial cost for the open-loop concept is less

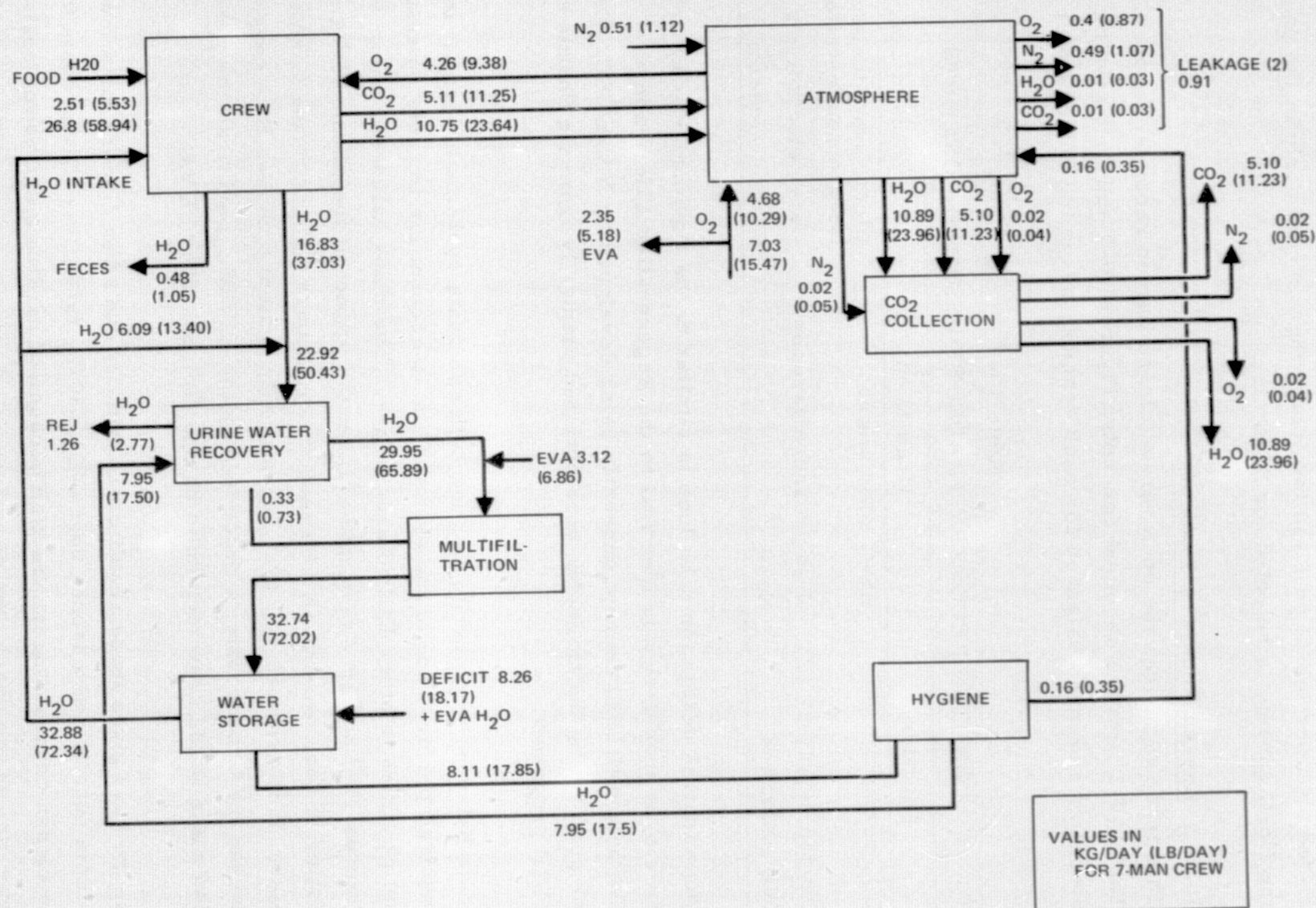


Figure 10-7. SCB Mass Balance Open O₂/Closed Water (7-Man Construction Shack)

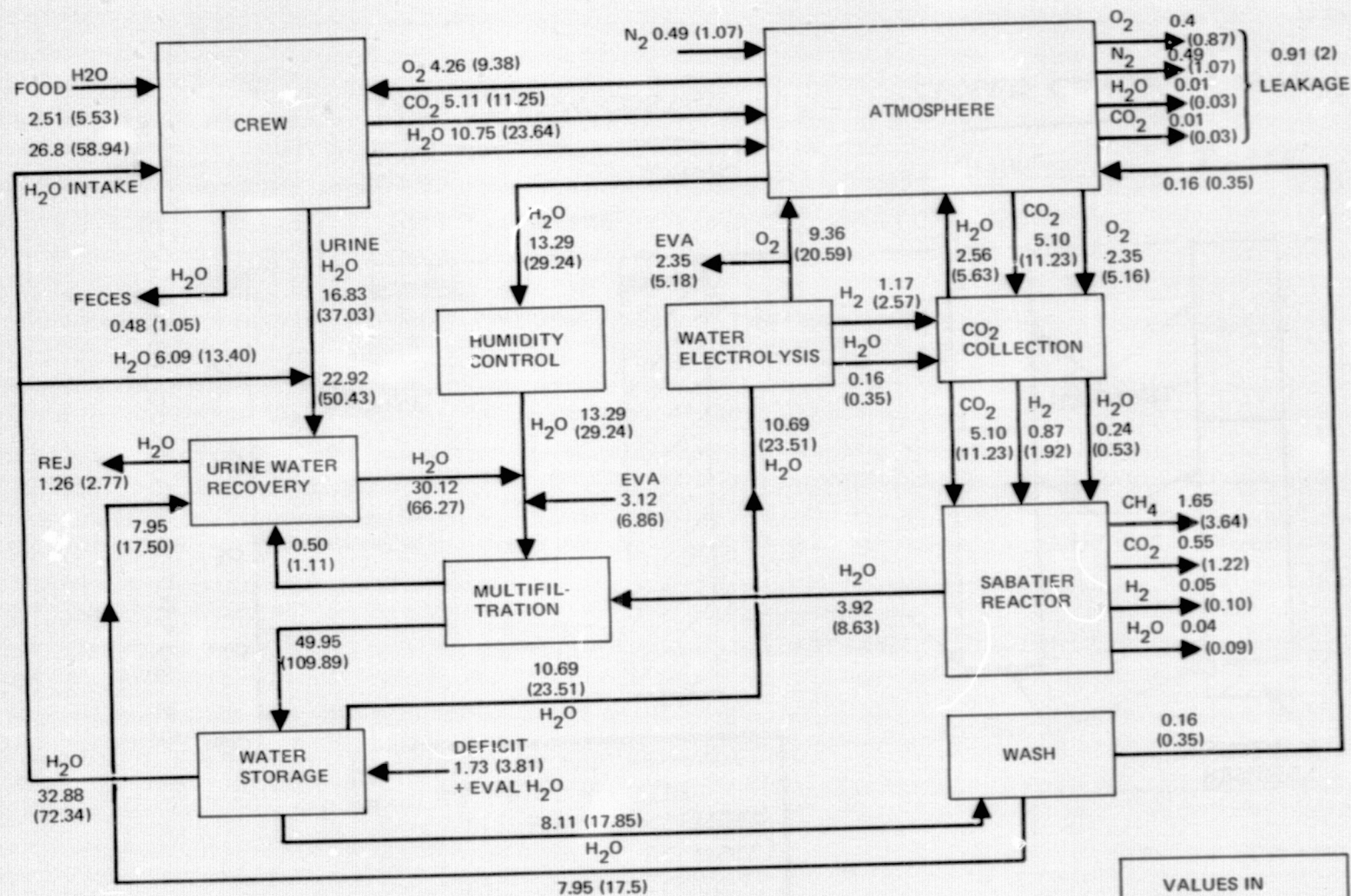


Figure 10-8. SCB Mass Balance Closed Oxygen/Closed Water (7-Man Construction Shack)

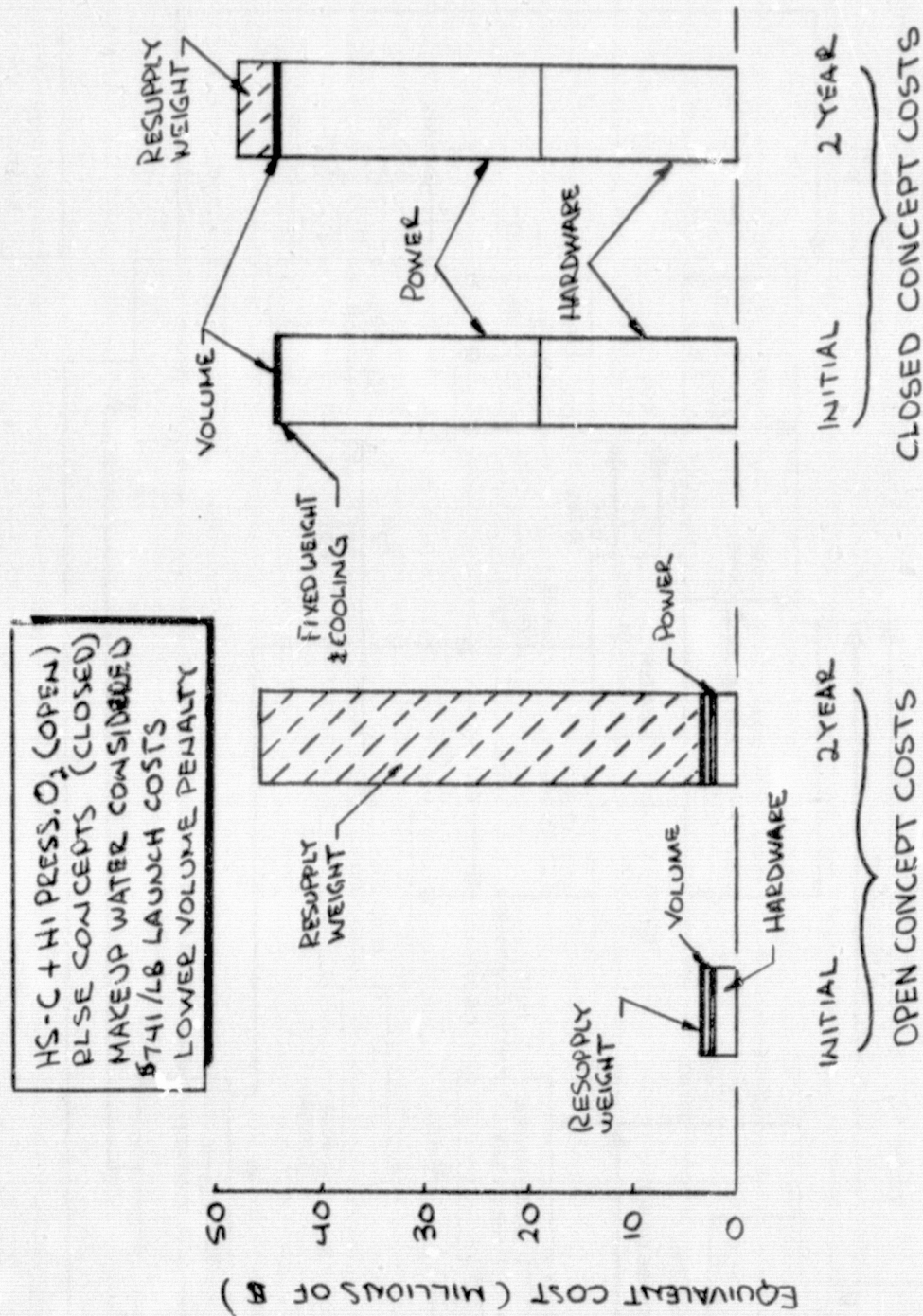


Figure 10-9. Oxygen Recovery Cost Trade

for HS-C because the humidity control assembly cost is saved. The cost of the closed O₂ concepts assume that a prototype has been built and tested and that all design and basic technical problems have been resolved.

Figure 10-10 shows the trade results as a function of mission duration, with and without power penalty and for varying launch costs. The results show that \$17 to \$41 million are saved initially with the open loop concept depending on power system costs. Oxygen recovery trades favorably early in the mission only at high launch costs or without power costs.

Based on the low initial cost and low early mission costs for low launch costs, open oxygen concept was selected initially. However, because the closed concept can potentially represent lower costs for long duration missions and is less reliant on resupply, it is recommended that the closed O₂ concept be retrofitted after the initial program phases.

10.3.4 Water Recovery Tradeoff

This trade compares water resupply with water recovery for supplying all or part of the water needs in the SCB. Potential water sources were assumed to be urine, urine flush, wash water, and multifiltration concentrate. Process rate for these sources is 31.2 kg/day (68.7 lb/day). A single unit is used to process all water sources, namely, vapor compression distillation. Wash water is not processed in a non-phase change process because this water source is small, 8 kg/day (17.5 lb/day), because wet wipes are used for personal hygiene rather than a whole body shower. This approach is consistent with an austere crew accommodation philosophy.

Water resupply (e.g., EMU cooling) is needed with or without water recovery and therefore tank development was not charged for either concept. However, costs associated with integration, subcontractor management and similar items were included.

It was also assumed that waste water must be returned to earth; not dumped overboard. The approach was taken wherein the same tanks supplying water were used to return waste water. As tanks are emptied, they are refilled

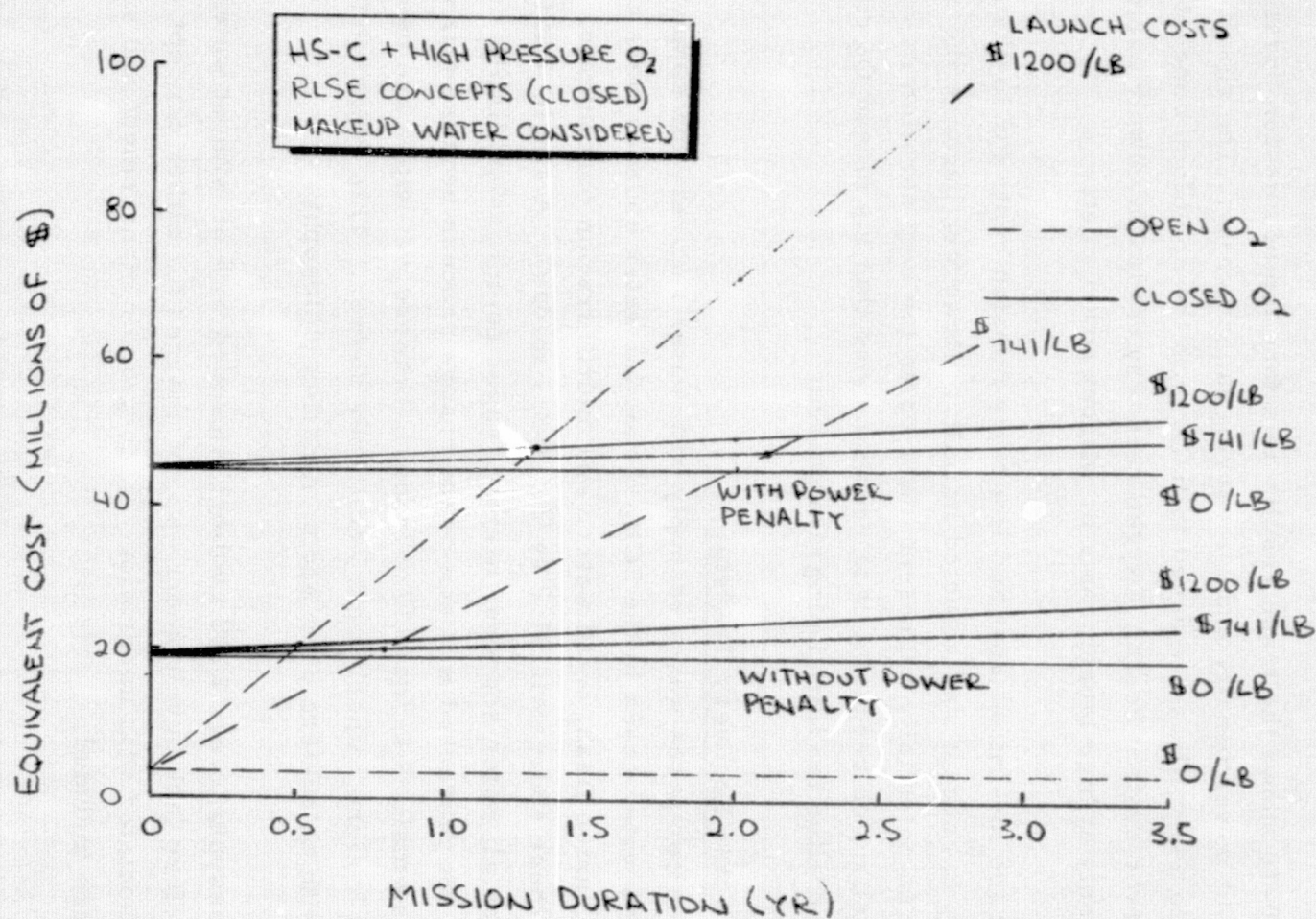


Figure 10-10. Open Versus Closed Oxygen Tradeoff

with waste water. An extra tank was supplied for waste water collection for the time before an empty tank becomes available. Also extra valves and lines were included to perform the operation. This approach may not be acceptable from a safety standpoint because of possible leakage between potable and waste water subsystems and/or because of the complexity of sterilizing the tanks on the ground prior to a resupply flight.

Results of the trade are shown in Figure 10-11 which indicates the major costs initially and after two years for a low volume penalty and a launch cost of \$1,630/kg (\$741/lb). Results show that the initial cost of open water concept is very low, but after two years the resupply weight results in a cost much higher than the closed water concept. Initial hardware cost is higher for water recovery but resupply is small.

Figure 10-12 shows the trade results in a form which clearly shows the effects of mission duration and launch costs. It can be seen that water recovery trades favorably for mission durations of less than a year except for low launch costs of below \$880/kg (\$400/lb). Water recovery will save substantial amounts of resupply, 14,000 kg/yr (31,000 lb/yr), and will reduce the dangers of handling waste water. Savings in the range of \$100 million can be expected in the first five years of operation. The above advantages for water recovery are believed to more than make up for the \$13 million higher initial cost. Therefore, water recovery using a single vapor compression distillation concept is selected for use on the low-cost SCB.

10.4 DISCUSSION OF RESULTS

The ECLSS concept trade results presented in this report support comparisons between ECLSS options, on a relative basis only. Current unknowns are associated with the trade factors resulting from operational considerations which may occur in the actual program planning such as: (1) available space may be available in the orbiter which could reduce chargeable resupply costs, (2) power scheduling can influence power system costs and (3) utilization of the crewtime could alter crewtime costs. Thus, the trades provide specific trends making it possible to identify and utilize the design drivers in concept evaluation.

VAPOR COMPRESSION
FOR RECOVERY (ELSE)
7 MAN UNITS
LOW VOLUME PENALTY
\$741/LB LAUNCH COST

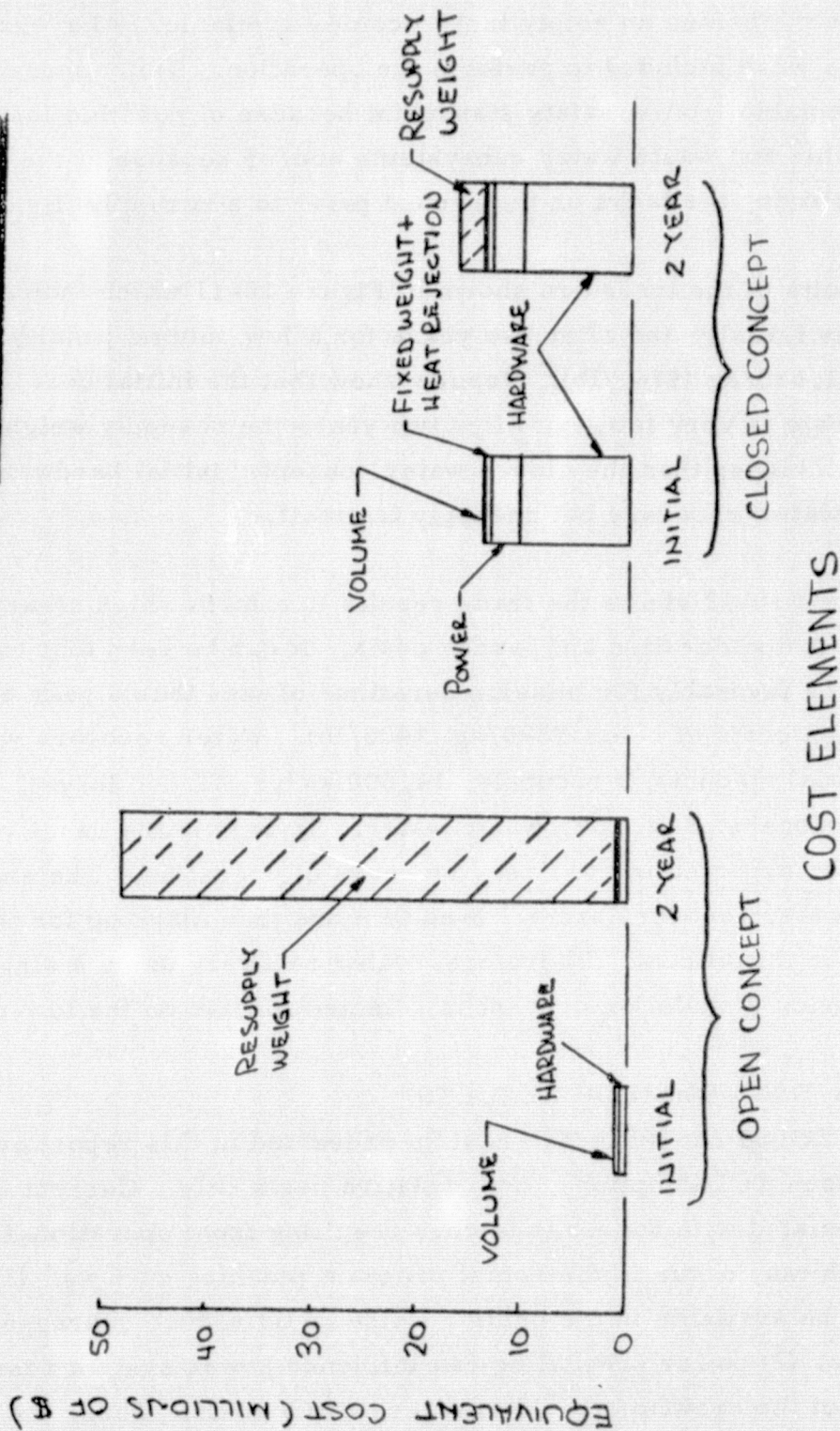


Figure 10-11. Water Recovery Tradeoff

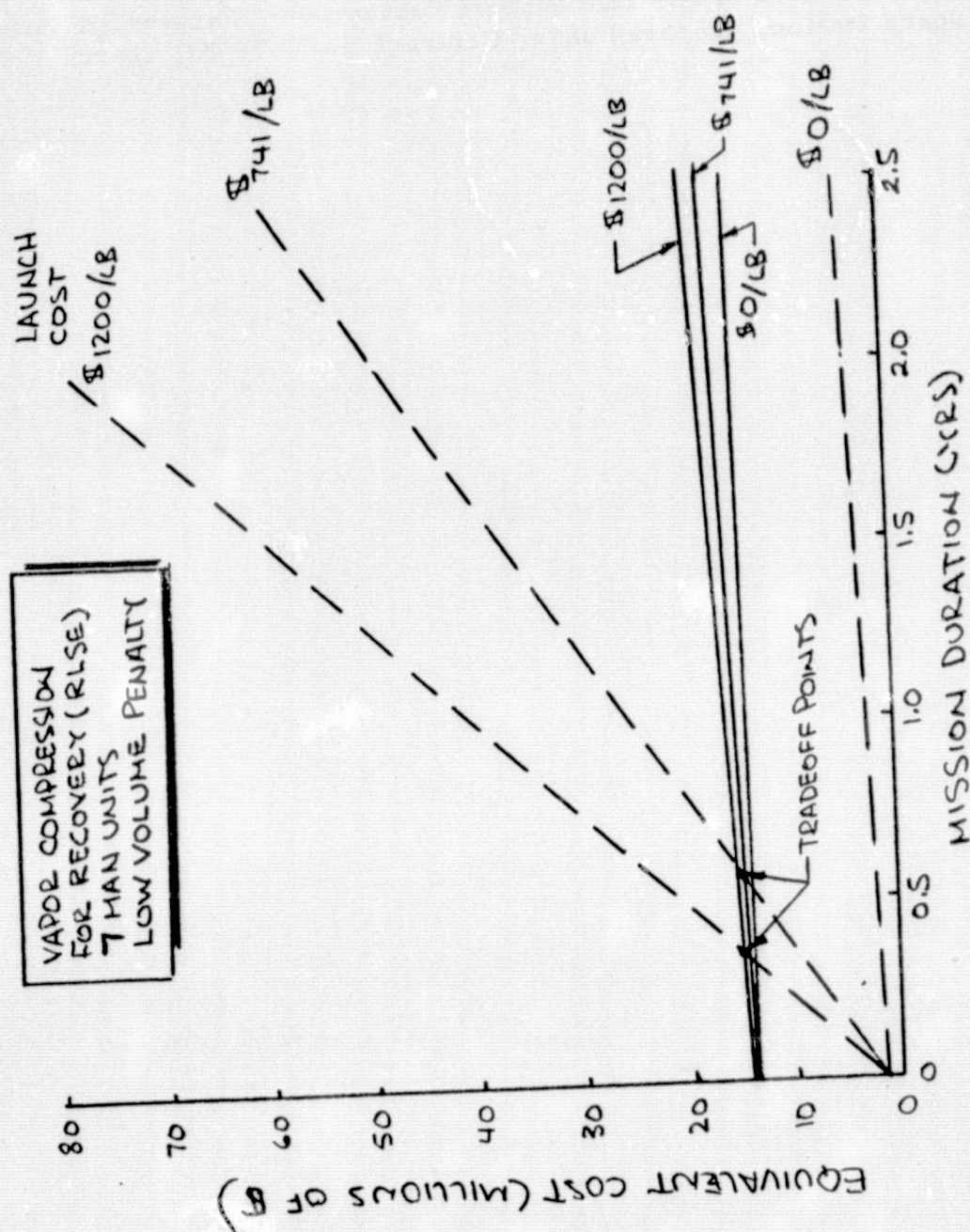


Figure 10-12. Water Recovery Tradeoff

10.5 REFERENCES

1. Low-Cost Space Construction Base Design Guidelines and Criteria, JSC-11867, June 1977.
2. Advanced Spacecraft Subsystem Cost Analysis, MDC E0027.
3. Thermal Control and Life Support Subsystems, Parametric Data for Space Station, prepared under Contract NAS8-14782, March 1977.



Section 11

THERMAL HEAT REJECTION ANALYSIS

11.1 INTRODUCTION AND SUMMARY

Radiator analyses were performed to assess the candidate configurations to allow a realistic comparison of program options. The analyses estimated the heat rejection for the primary types of radiator designs, and orientations considered in the study. Specifically the following cases were considered.

- A. Integral (cylindrical) radiators
 - 1. Two orientations
 - 2. Low temperature and high temperature (space processing)
 - 3. Variable radiator outlet temperatures
- B. Deployed radiators (flat plates)
 - 1. Two orientations
 - 2. Variation of surface characteristics
 - 3. Variable radiator outlet temperatures
- C. Power module radiator

A fundamental closed-form solution based on the sink temperature concept for accounting for heat influx to the radiators was used. This approach is valid for steady-state, flat-plate radiator configurations.

Results of the analysis indicate that for the estimated heat loads, considerable excess heat rejection capacity exists for the radiator configurations analyzed. The solution used the assumptions of steady state and heat influx from adjacent vehicle elements such as solar arrays and other modules. However, degraded and unsophisticated coating characteristics were assumed to assure that the analytical results give a conservative first assessment of SCB radiators.

11.2 ANALYSIS DESCRIPTION

This section describes the purpose, boundary conditions, assumptions, and technical approach to the analysis. The analytical approach used was based

largely on that described in Reference 1. This approach was appropriate because of the several SCB configurations, radiator arrangements, and orbital orientations which had to be analyzed during Part 3. The results obtained useful in a Phase A-level evaluation of heat rejection concepts.

11.2.1 Purpose

The radiator performance analytical task assessed the various SCB configurations associated with evaluation of program options. Specifically, the heat rejection capability was estimated for key configurations and compared with requirements. Both integral (cylindrical) and deployable (flat plate) geometries were considered, as were different orientations. Results were used to recommend modifications in configuration arrangements and orientations. Additionally, instances were identified where integral radiators were deficient and deployable radiators must be considered. The option of using deployable radiators only was also considered.

Determination of key impacts on heat rejection concepts relative to the desirability and cost of the program options was the key objective of the analysis. This allowed a complete and reasonably precise comparison and evaluation of options.

11.2.2 Boundary Conditions and Assumptions

The analysis performed was based on certain boundary conditions and assumptions; these were established to approximate anticipated physical conditions or to simplify the configuration comparison.

11.2.2.1 Environment

Heat influx to the radiator surfaces were assumed to be:

- A. Direct solar
- B. Earth albedo
- C. Earth infrared (IR)

IR from another vehicle elements was considered only in one case, which analyzed a deployable radiator attached to the underside of a solar power module. This analysis assumed no reradiation between radiator and solar

array. A summary list of the environmental values used is presented in Table 11-1.

11.2.2.2 Radiator Characteristics and Orientations

Deployed radiators were assumed to be two flat plates, each with separate tubes and the two radiator sides were insulated from one another. A single surface and tube network performance would be nearly the same as the model analyzed.

Two orientations were considered for deployed radiators as shown in Figure 11-1. The orientation termed perpendicular to the orbit plane always has the same side facing the earth surface. The orientation termed parallel to the orbit plane always has the same edge facing the earth's surface.

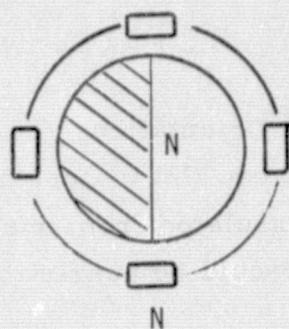
Two orientations were analyzed for the integral (cylindrical) radiator; (1) vehicle centerline parallel to the velocity vector and (2) vehicle centerline perpendicular to the orbit plane. In both orientations, radiator tube layout was assumed to consist of two panels which allowed flow away from the earth.

Table 11-1
ENVIRONMENTAL ASSUMPTIONS

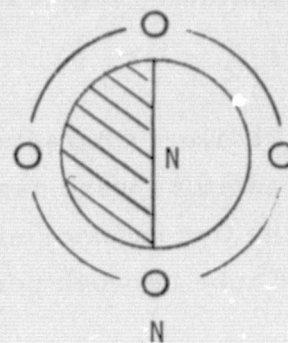
Item	Value
• Solar constant	— 1.395 kW/hr m ² (443 Btu/hr ft ²)
• Earth albedo	— 0.35
• Altitude	— 456 km (246 nautical miles)
• Solar array reflectivity	— 0.19
• Solar array absorptivity (solar wavelength)	— 0.81
• Emissivity (IR)	— 0.81
• Solar array temperature - sun	— 80°C (176°F)
• Solar array temperature - shade	— -80°C (-112°F)
• Angle between solar vector and orbit plane (β)	— 0

TOP VIEW

SIDE VIEW



ϕ ALONG VELOCITY VECTOR

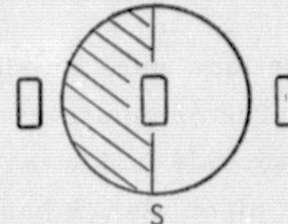
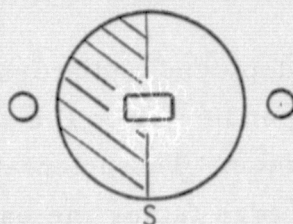
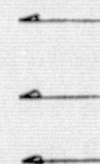


$\phi \perp$ TO ORBIT PLANE

CR60



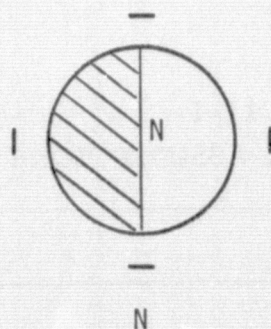
SOLAR



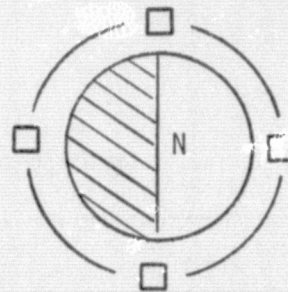
INTEGRAL (CYLINDRICAL)

TOP VIEW

SIDE VIEW

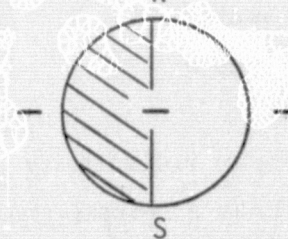
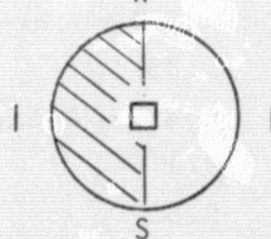
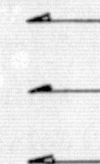


$\perp$ TO ORBIT PLANE



$\parallel$ TO ORBIT PLANE

SOLAR



DEPLOYED (FLAT PLATE)

Figure 11-1. Radiator Orientations

This arrangement is the best flow pattern for shade-side operation. See Figure 11-1 for clarification.

The radiator assumed for the 38 kW Configuration power module, shown in Figure 11-2, is a flat plate perpendicular to the solar array. The orientation is with the solar arrays facing the sun direction for all points in the orbit; the radiator is located on the side opposite the sun.

11.2.2.3 Geometry of Thermal Models

Figure 11-2 shows the thermal models for the cylindrical radiator and the solar power module radiator. Single flat-plate models are not shown as their geometry is obvious.

The cylinder was divided into 12 flat plate sides so that the closed form solution could be used. Radiator fluid flow was assumed to flow from Surface 1 to Surface 6 around both sides of the vehicle. The fluid conditions out of Surface 1 were assumed to be the inlet fluid conditions for the next surface, for instance, surface 2. This assumption was necessary to simulate circumferential fluid flow. A cylinder model length was not assumed; the analysis was performed parametrically, based on radiator area divided by fluid heat capacity rate (A_R/WC_p).

Geometry of the power module radiator is also shown; the radiator is small compared to the size of the solar arrays.

11.2.3 Technical Approach

The technical approach was based on the well-known exact solution analysis reported on in Reference 1. This solution is for a flat plate, with the environment represented by an effective sink temperature defined as follows:

$$T_s = \left\{ \frac{1}{\sigma} \left[\frac{\alpha_s}{t} S F_s + \frac{\alpha_s}{\epsilon_t} a S F_{E(SR)} + \frac{(1-a)}{4} S F_{ET} \right] \right\}^{1/4}$$

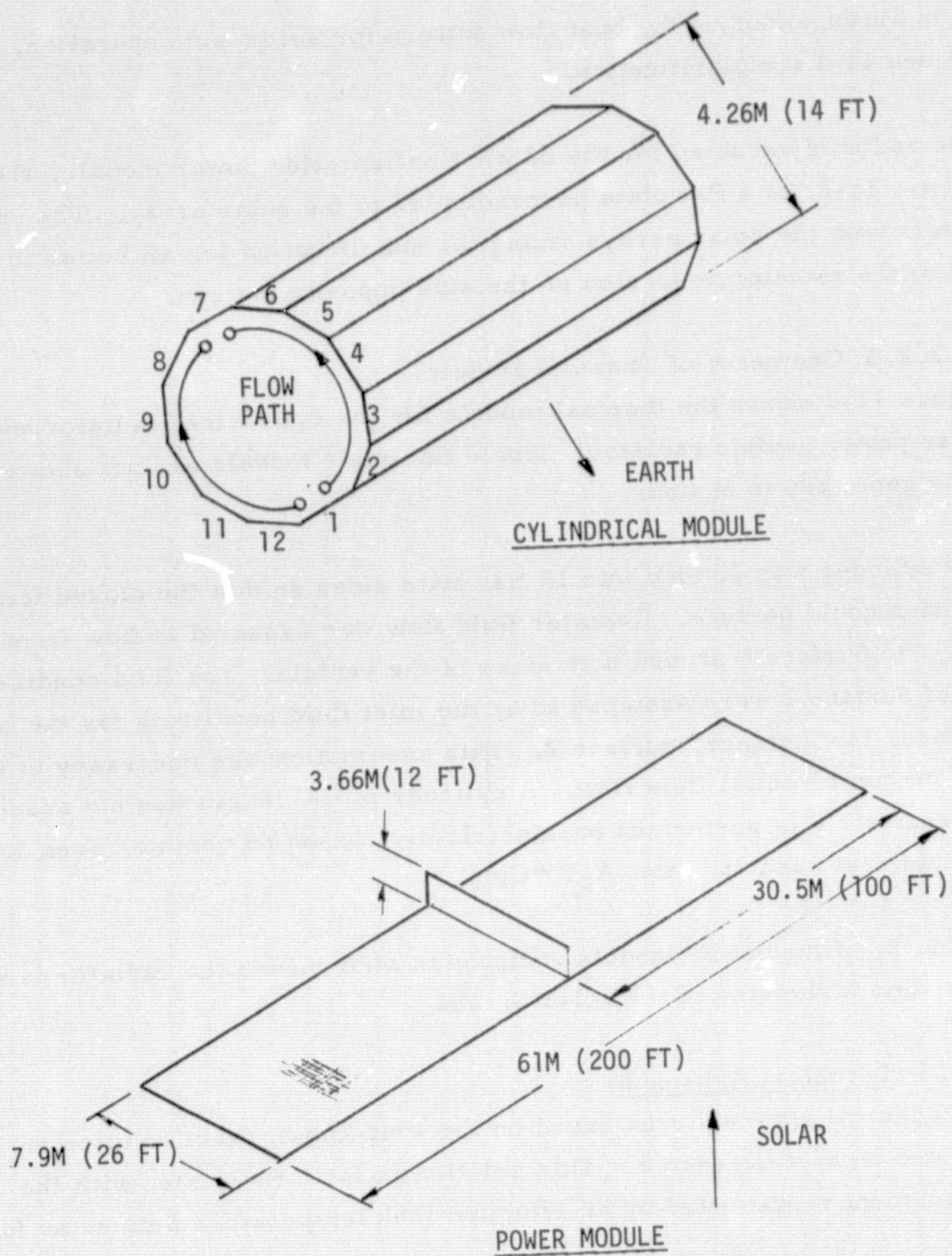


Figure 11-2. Radiator Thermal Model Geometry

where:

T_s - Sink temperature

α_s - Absorptivity in solar wavelength

S - Solar constant

F_s - View factor, solar

a - Earth albedo

$F_{E(SR)}$ - View factor, solar reflected

F_{ET} - View factor, earth thermal

Values for the various view factors were obtained from Reference 1 for different orbital locations and orientations.

The sink temperature approach allows a basic equation to be written for heat transfer from an increment of radiator area.

$$dq_{R, \text{ net}} = \sigma \epsilon_t (T^4 - T_s^4) dA$$

where,

T - Temperature of increment

By establishing the heat transfer from the radiator increment equal to heat transferred from the radiator fluid in the increment, and integrating the resulting equation, a closed-form solution can be obtained.

$$\zeta(\tau_1) - \zeta(\tau_2) = \frac{\sigma \epsilon_t \eta_o T_s^3 A_R}{W \dot{C}_p} \quad (2)$$

where:

$\tau = T_R/T_s$ = Temperature Radiator/Temperature Sink

σ - Stefan - Ballzman constant

ϵ_t - Emmissivity (thermal)

η_o - Fin effectiveness

A_R - Radiator area

W - Fluid flow rate
Cp - Specific heat

$$\zeta(\tau) = \frac{1}{4} \ln \left[\frac{\tau + 1}{\tau - 1} \right] + \frac{1}{2} \tan^{-1} \tau \quad (3)$$

These equations require an iterative solution wherein Equation (2) must be solved for $\zeta(\tau_2)$; all other terms are known. In simple terms, the solution must be found for

$$\zeta(\tau_2) = \text{constant}$$

This requires an iterative solution since τ , Equation (3), cannot be solved in closed form.

Both heating and cooling conditions were obtained by use of this solution; however, absolute values of the ln term must be used in the heating case. Additionally, the following modified solution was used for cases where the sink temperature was zero:

$$T_{R2} = \left[\frac{T_{R1}^3 W C_p}{3 \sigma \epsilon_t \eta_o A_R T_{R1}^3 + W C_p} \right]^{1/3} \quad (4)$$

As mentioned previously, the solution is for steady state; however, a prediction of performance was obtained by the average performance for 12 points around the orbit. A constant radiator inlet temperature was used for all points in the orbit. The parameter A_R/WC_p was varied to obtain a range of radiator outlet temperatures. Heat rejected from the radiator was calculated based on radiator flow rate and temperature drop.

11.3 RESULTS OF ANALYSES

Results are presented in this section, primarily in parametric format, as a function of radiator outlet temperature. Both integral (cylindrical) and

flat-plate (deployable) types of radiator data are included. Effects of inlet temperature, radiator orientation, and surface characteristics are included.

11.3.1 Integral Radiator Performance

Figure 11-3 shows the performance for integral radiators. Performance is better for a vehicle oriented along the velocity vector because less direct solar impinges on the radiator surface. Because of this, configurations which orient cylinder centerline perpendicular to the orbit plane are to be avoided for low beta angles. This effect reverses at large beta angles because then solar impingement becomes larger for the orientation with vehicle centerline oriented along flight vector.

Variation of performance with radiator outlet temperature is not large, $0.00145 \text{ kW/m}^2\text{°C}$ ($0.266 \text{ Btu/hr ft}^2\text{/°F}$), because of the averaging effect of the wide variety of environments seen by the radiator surfaces.

Normal ECLSS outlet temperatures are expected to be about 2°C (35°F) for condensing humidity control and 13°C (55°F) for the initial baseline where the HS-C CO_2 control unit controls humidity. Performance range for these temperatures is about 0.122 kW/m^2 (39 Btu/hr ft^2) to 0.106 kW/m^2 (34.3 Btu/hr ft^2). This is based on a degraded coating of $\alpha/\epsilon = 0.39$. Performance early in the mission is expected to be somewhat better.

11.3.2 Deployable Radiator Performance

Performances for flat-plate radiators (deployable) are shown in Figures 11-4 and 11-5 as a function of radiator outlet temperature. An orientation placing the flat plate parallel to the orbit plane, Figure 11-4, shows high heat rejection capability, amounting to between four and five times greater than for a cylindrical-shaped radiator at a 12.8°C (55°F) outlet temperature. Half of this increase is due to the two-sided heat rejection from the flat plate.

Additional performance gains are derived because the deployable radiator (1) has no direct solar impingement and (2) has a relatively low view factor for earth albedo and IR. A drawback to this orientation is that the radiator never has direct view of outer space. Although performance is very high at low beta angles, at higher beta angles, performance is expected to decrease because of greater solar energy impingement, especially with high solar coating absorptivities.

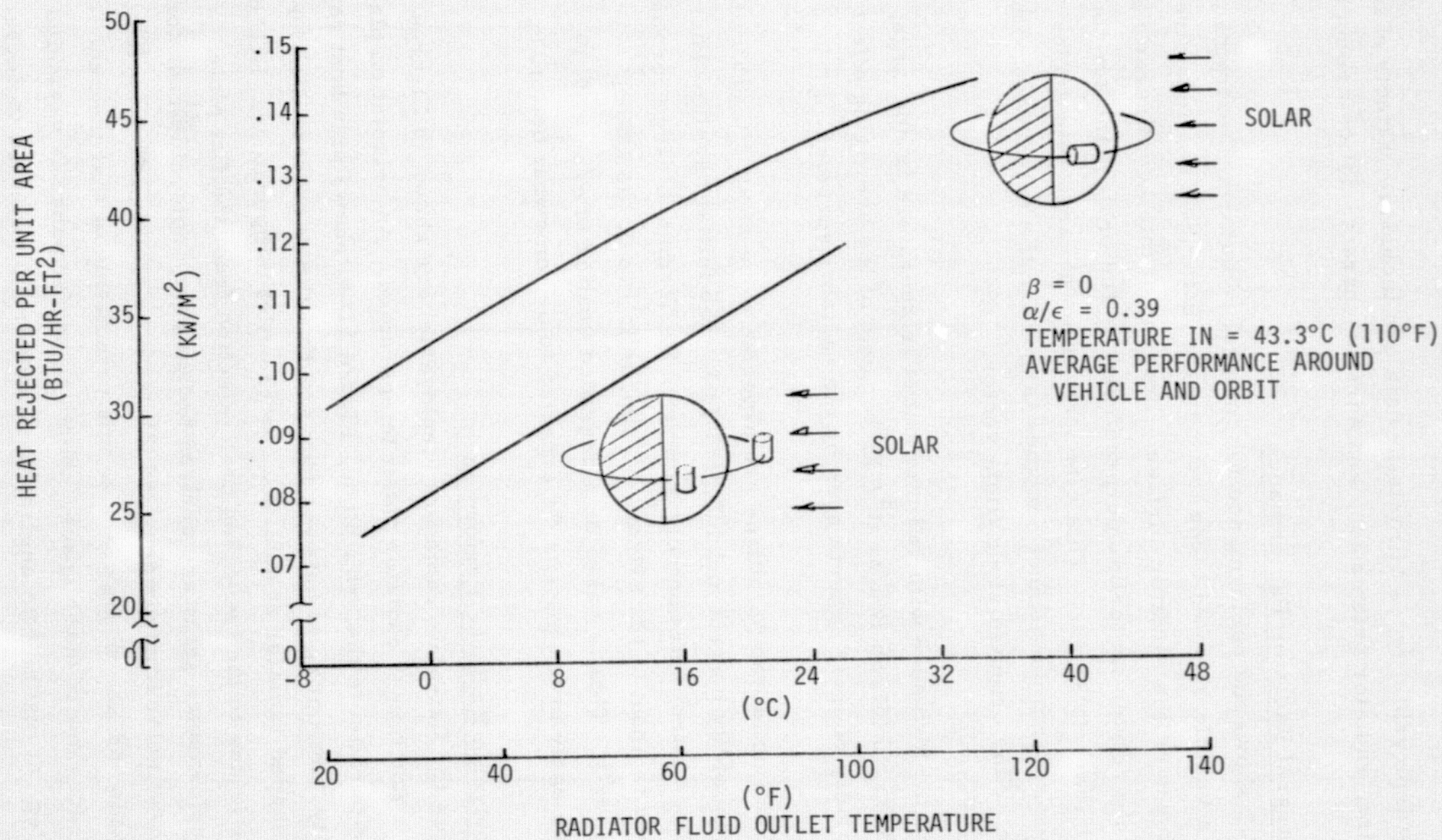


Figure 11-3. Radiator Performance for Integral Cylindrical Configuration

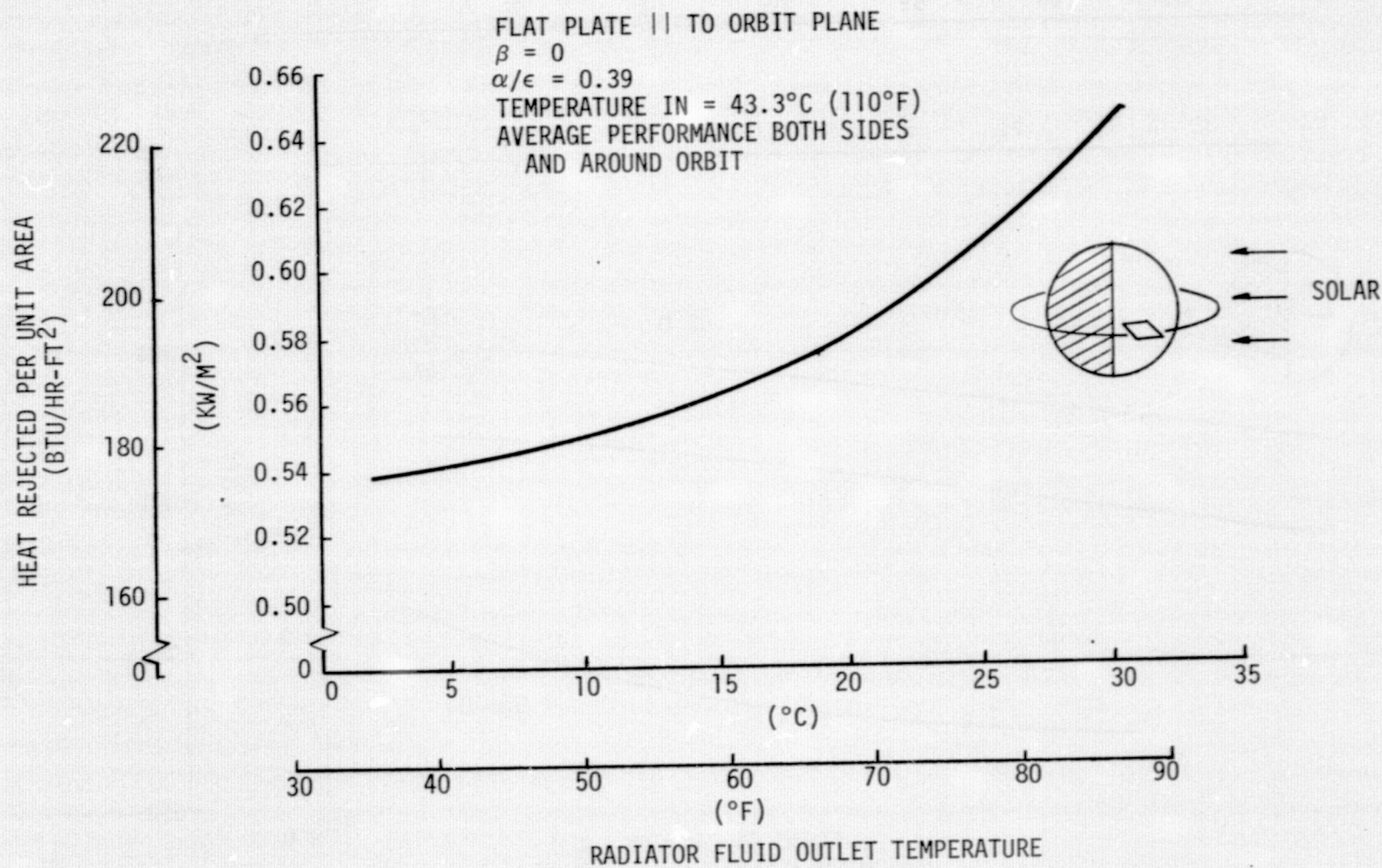


Figure 11-4. Deployable Radiator Performance — Variation with Outlet Temperature

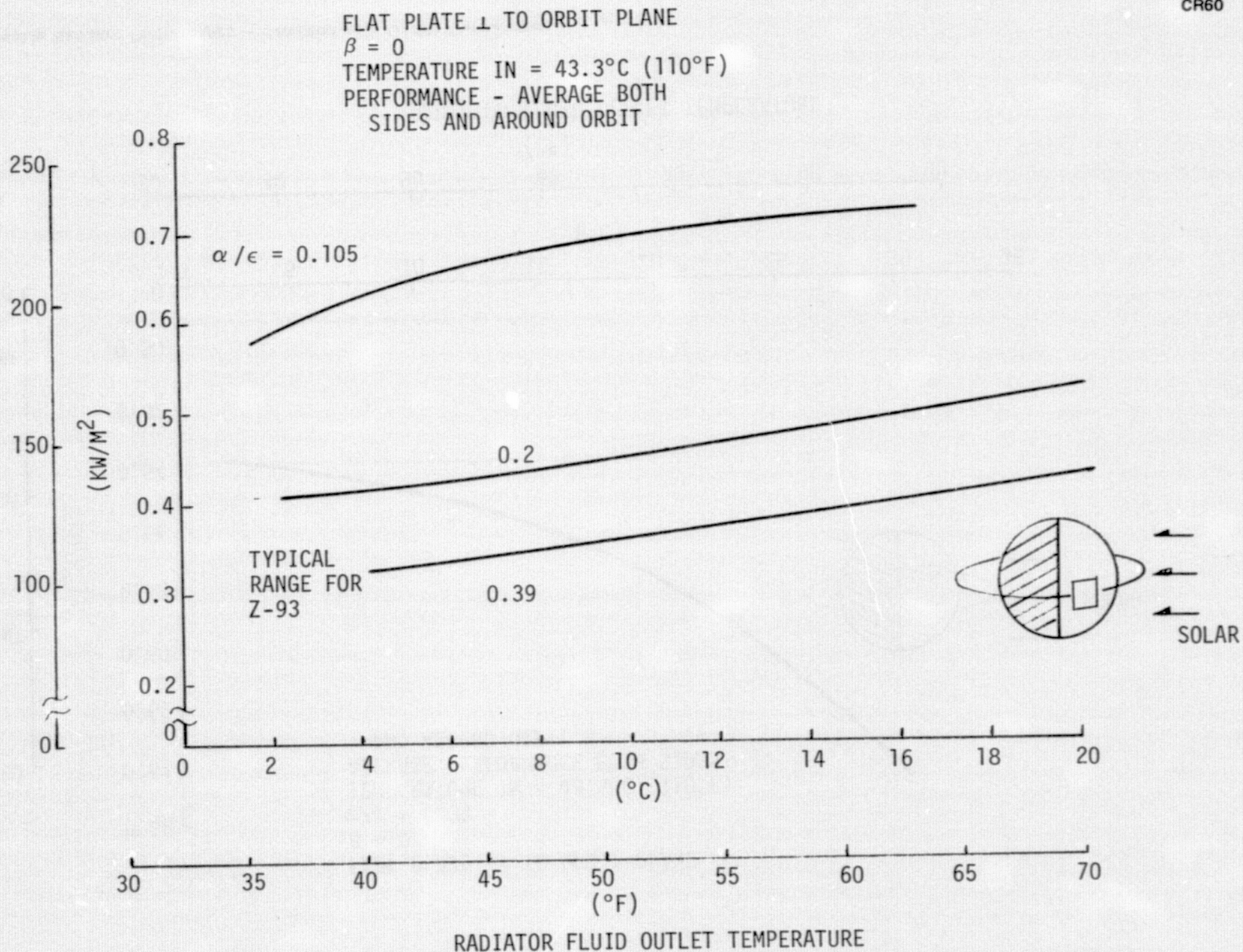
HEAT REJECTION PER UNIT AREA
(BTU/HR-FT²)

Figure 11-5. Deployable Radiator Performance - Variation with Outlet Temperature

Figure 11-5 shows flat-plate heat rejection for an orientation perpendicular to the orbit plane. Performance for degraded coatings, high solar absorptivity, is lower than for the orientation discussed previously because of large solar impingement and earth albedo on the sun side of the orbit. Additionally, the radiator side facing the earth's surface receives large earth IR impingement. This effect is offset somewhat because the surface facing away from the earth faces deep space with a low heat influx during a large portion of the orbit. At a radiator outlet temperature of 12.8°C (55°F) the radiator parallel to the orbit plane rejects about 1.5 times as much heat as the radiator perpendicular to orbit plane. At lower radiator outlet temperatures, the radiator parallel to the orbit plane performs better, because a lower sink temperature exists for this case. The radiator perpendicular to the orbit plane is expected to be much more sensitive to coating solar absorptivity, and at very low values, the performance is very high. Also the performance at this orientation improves as beta angle increases because solar impingement and earth albedo heat influxes decrease.

11.3.3 Space Processing Module Heat Rejection

Space processing module radiator characteristics differ considerably from the space construction module and construction shack because higher fluid temperatures are expected. Figure 11-6 shows results of analyses for 260°C (500°F) inlet temperature and a variable outlet temperature. Two types of radiators are considered for space processing modules, i. e., single and dual. The single radiator concept uses a single radiator for high-temperature processing loads and normal ECLSS loads, such as air cooling and conventional avionics cooling. This concept would require a relatively low radiator outlet temperature of 1.7°C (35°F) to 12.8°C (55°F). Performance for the single radiator concept, based on 260°C (500°F) inlet, would be about 0.5 kW/m² (159 Btu/hr ft²) for the integral radiator and 1.05 kW/m² (333 Btu/hr ft²) for the deployed radiator.

The dual-radiator concept uses a low-temperature radiator for the relatively low temperature and heat loads from the ECLSS, and a separate, high-temperature radiator for the large heat loads at high temperature associated with space processing equipment. This approach results in a lower overall

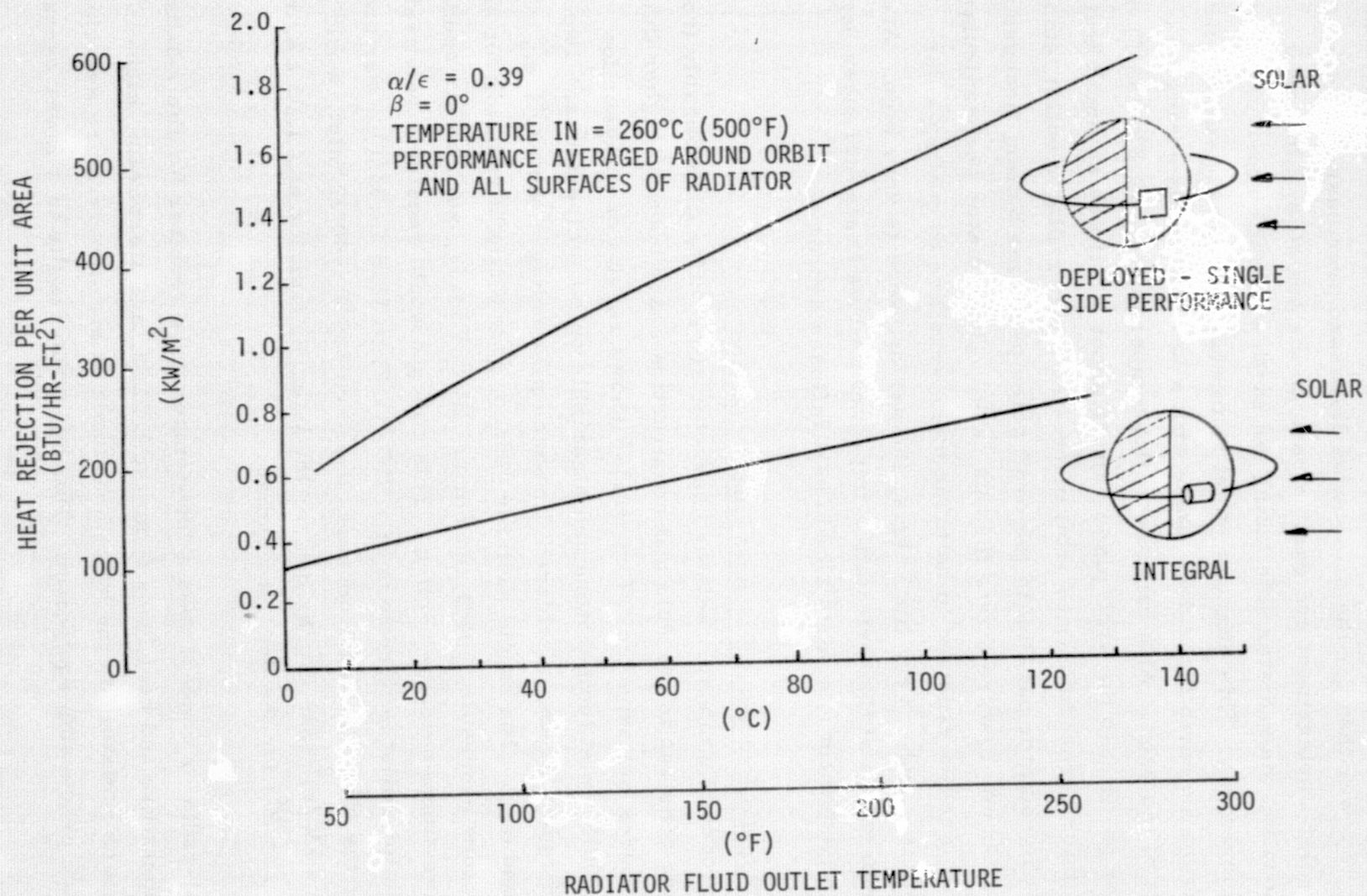


Figure 11-6. Space Processing Module Heat Rejection

radiator area but adds additional equipment for two separate thermal control loops. This approach also allows the use of a high-temperature fluid for the space processing radiator; the normal Freon 21 radiator fluid has a high vapor pressure at high temperatures. Use of the separate radiators can reduce the size of the SPM radiator by 1/2 to 1/3 times.

11.3.4 Power Module Heat Rejection

Power module deployable radiator heat rejection was determined for the geometry given in Figure 11-2. The results showed that about 10.2 kW could be rejected with a 78.9 C (110°F) inlet temperature and a 12.8 C (55°F) outlet temperature. The analysis assumed constant array temperatures for the shade side and sun side, that no heat flux from the radiator reached the array, and that only IR heat flux reach the radiator from the array. The model represents only one-fourth of the total radiator on the SEPS-based PM.

11.3.5 Comparison of Results with Requirements

Requirements for heat rejection were determined and are shown in Table 11-2 along with performance capability as determined from the paragraphs above.

Table 11-2
HEAT REJECTION REQUIREMENTS AND CAPABILITY

Vehicle Element	Requirement (kW)	Capability (kW)
Construction Shack	8	17.8 ⁽¹⁾
Space Construction Module	3	7.7 ⁽¹⁾
Space Processing Module	8 to 15	12 to 34 ⁽²⁾
Power Module	19 to 49 ⁽³⁾	41

(1) Integral radiator oriented along velocity vector and degraded coating.

(2) Low value for single radiator, high value for dual radiators with 20% of load on low-temperature radiator.

(3) 19 kW is load for power module only, 49 kW rejects all heat generated. Power module produces 38 kW at bus, beginning of life.

The results show that the radiator capability surpasses requirements for the construction shack and space construction module. Excess capability of about 100 percent exists for these modules. The space processing module should have excess capability if a dual-radiator system is used to take advantage of the high efficiency of high-temperature radiators. Sufficient capacity exists in the power module to reject heat attributed to power module equipment plus about 75 percent of the power at the bus. The analyses performed assumed steady state, zero thermal mass, and in most cases no heat influx from adjacent vehicle elements. However, sufficient excess heat rejection capacity exists in the selected SCB configuration to assume that the radiator concepts analyzed possess adequate performance to meet the operational requirements.

11.4 REFERENCES

1. A. F. Anderson et al. Radiator Design for Space Vehicles. Garrett Corporation.
2. J. A. Stevenson and J. C. Grafton. Radiation Heat-Transfer Analysis for Space Vehicles, ADS61-119. USAF. 1964.
3. Modular Space Station Detailed Preliminary Design, MDC G2585 - Vol II. McDonnell Douglas Astronautics Company. 1971.
4. A. J. Chapman. Heat Transfer. The Macmillan Company, 1960.

SECTION 12

SPACE RADIATION EFFECTS ANALYSIS

The Space Radiation Effects Analysis included the definition and analysis of the radiation environment, the formulation of man and configuration models, determination of in-module and EVA dose received, and analysis of overall shield requirements. These items are discussed below and also summarized in Section 6 of the Technical Report.

12.1 Environment Definition/Allowable Dose

The low earth orbit mission regime was of prime interest for the Space Station Programs studied in Part 3. Accordingly, the space radiation environment was defined for orbits in the 28.5 to 55° inclination, 400 to 500 KM altitude range.

The environment models used for this analysis are listed in Table 12-1.

TABLE 12-1

ENVIRONMENT DEFINITION

<u>Radiation Source</u>	<u>Model</u>
Electrons	
Inner Zone	AE-6 (Solar Max.) AE-5 (Solar Min.)
Outer Zone	AE-7 (High & Low)
Protons	
0.1-4 MeV	AP-5
4 - 30 MeV	AP-6
30-50 MeV	AP-6 (Extrapolated to 30-50 MeV)
> 50 MeV	AP-7
SCR	November 12, 1960 Flare

AE-5 was used for the inner zone trapped electron definition since it produced the higher radiation level and the 1985 era (solar min.) was of major

interest. The high value model of AE-7 was selected for the outer zone electrons.

AP-6 was extrapolated into the 30-50 MeV energy range to complete the trapped proton spectrum. The AP-6 extrapolation was used to avoid discontinuities present with the use of AP-1. AP-3 was later supplied as the integrated proton model to be used for future analyses.

The models used were obtained from the National Space Science Data Center, GSFC, in tape format. An included program (TRECO) was used along with MDAC programs CHARGE, CAMERA, and SIGMA for the data analysis. The Solar Cosmic Ray (SCR) model used was the November 12, 1960 event as specified by JSC. A statistical flare model was also analyzed for comparative purposes.

The allowable dose criteria used are listed in Table 12-2.

TABLE 12-2
RADIATION EXPOSURE LIMITS

Constraint	Primary Reference Risk	Ancillary Reference Risks		
	(REM at 5 CM)	Bone Marrow (REM at 5 CM)	Skin (REM at 0.1 MM)	Ocular Lens (REM at 3 MM)
1-Year Average Daily Rate		0.2	0.6	0.3
30-Day Maximum		25	75	37
Quarterly Max. ^A		35	105	52
Yearly Maximum		75	225	112
Career Limit	400	400	1200	600

^A May be allowed for two consecutive quarters followed by six months of restriction from further exposure to maintain yearly limit.

These were obtained from the NASA Space Science Report of 1970 - "Radiation Protection Guides and Constraints for Space-Mission and Vehicle-Design Studies Involving Nuclear Systems." The primary reference risk dose corresponds to an added probability of radiation - induced neoplasia over a period of about 20 years that is equal to the natural probability for the specific population at risk. The ancillary risk levels are calculated from the primary. In practice for the missions at hand, the skin and eye dose are the limiting factors in the three months to six months time range. Further, the skin was used for most of the calculations since it is the most difficult against which to shield.

12.2 Configuration and Man Model

The man dose calculations were determined both for a man model residing within the Space Station and for an EVA crewman in an EVA suit.

The Space Station configuration was and is an evolving concept whose overall configuration will vary as more study is accomplished and even as the mission progresses. The approach used was then to consider the man dose received within a typical habitable module without regard to the surrounding modules as potential shadow shields with the exception of the module ends. The module ends were assumed thick as they would be if docked to another. Figure 12-1 shows a typical module (16 M long x 4.4 M diameter). The cylinder wall construction would typically be as in Figure 12-2 as developed in the 1969 Phase B Space Station Study. The minimum path thickness is about 0.1" (0.7 gm/cm^2). Radiation, waffle shoulders, equipment, etc., add to the average thickness but are usually not well distributed. Detailed past studies have found that these do not contribute greatly to the overall shield unless they are near uniform distribution, therefore, the minimum path thickness was used. Of course, the thickness varies with the cosine of the ray incidence angle and is accounted for in the geometry computer programs. The thickness was allowed to vary to attenuate the radiation dose as needed. The right hand portion of Figure 12-1 was assumed to be a typical 7-man Biowell location if needed for near continuous short residency during a solar cosmic ray event. Heavier shielding was added as needed to this area.

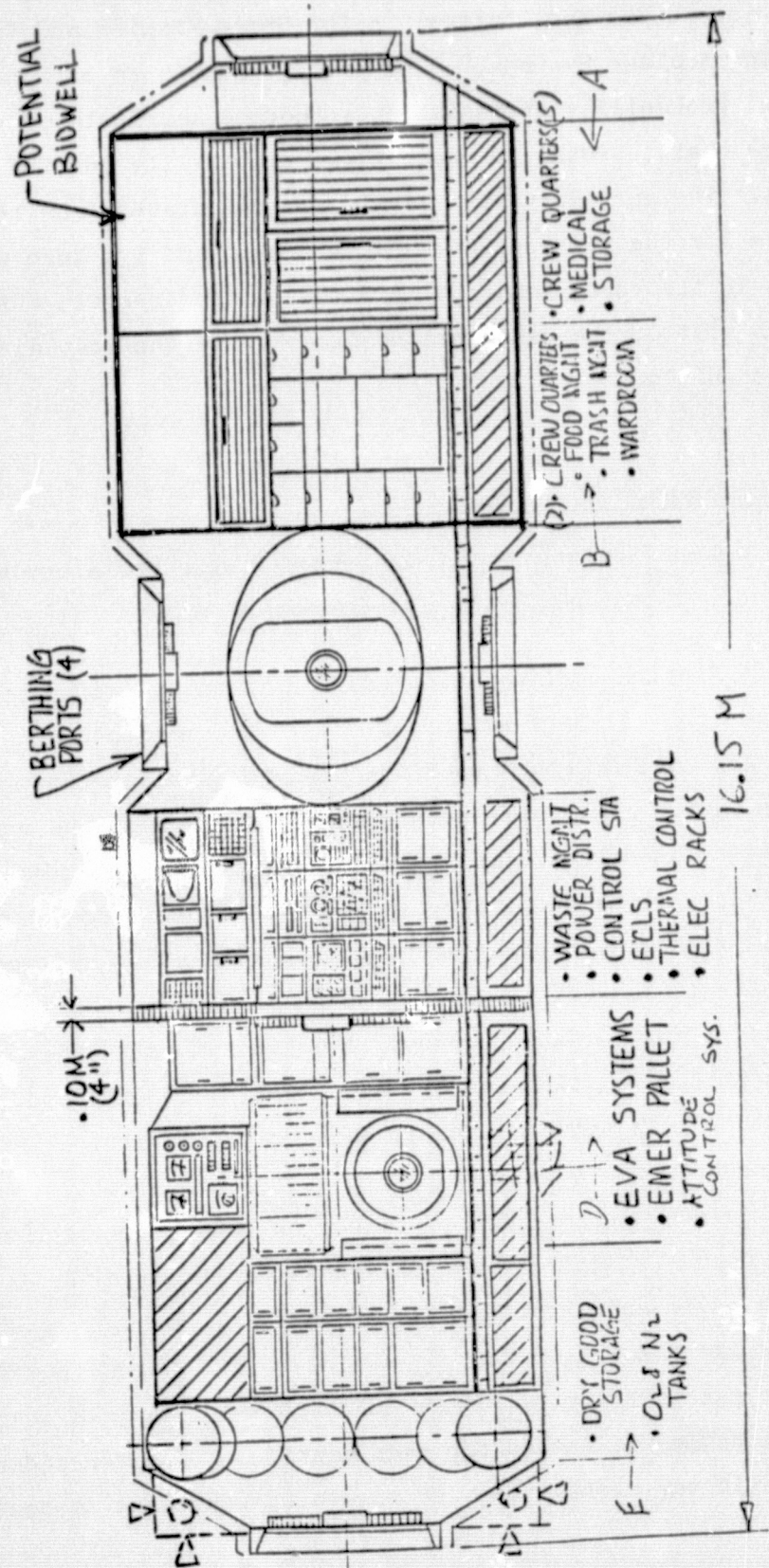


Figure 12-1. Space Station Module

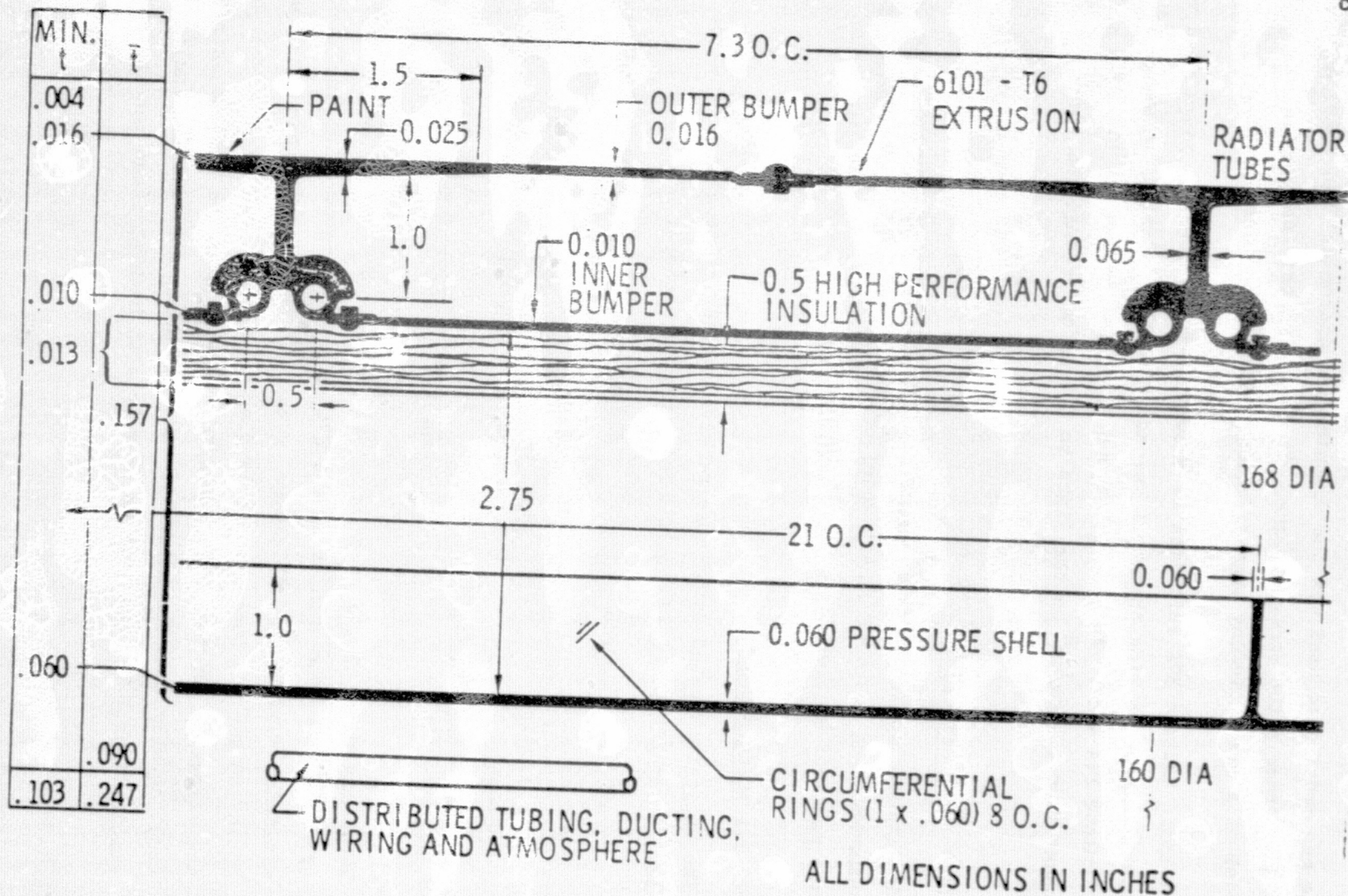


Figure 12-2. Structural Details

The man model data used was the Computerized Anatomical Man (CAM) developed by MDAC under prior contract work. It is a geometric and anatomical detailed model of a man including organs, voids, bones, bone marrow, etc.

The basic Shuttle EVA suit data is shown in Figure 12-3. A simplified version (spherical) of varying thickness was added to the detailed CAM model to calculate EVA dose. As seen in Figure 12-3, the equivalent EVA suit shield ranges from 0.1 gm/cm^2 on the limbs to 0.4 gm/cm^2 around the torso.

12.3 Dose Calculations

Radiation dose to the skin, eyes, and BFO (blood forming organ) were calculated using the environment model of Section 12.1 and the module and man model of Section 12.2.

12.3.1 Module Dose

The orbit integrated daily dose received while inside a Space Station module is listed in Table 12-3 for the following parameters;

Environment Source	Electrons, Bremsstrahlung, Protons, SCR				
Altitude	400, 450, 500 KM				
Inclination	28.5°, 48°, 55°				
Receptor	Skin, Eyes, BFO				
Module Wall Thickness	In.	.075	.1	.15	.20
	gm/cm^2	.514	.686	1.03	1.37

Data from these tables were summed to form the trapped radiation dose plots of Figure 12-4 for skin, eye, and BFO as a function of orbit inclination and altitude. Module wall thickness is 0.1 in. The daily allowable doses for 90 to 180 day missions are overlaid as shown. The BFO dose is well below the allowable for the orbit conditions used. Both the eye and skin dose are about equally near the allowable for some upper altitude and inclination combinations. The dose varies sharply with both altitude and inclination.

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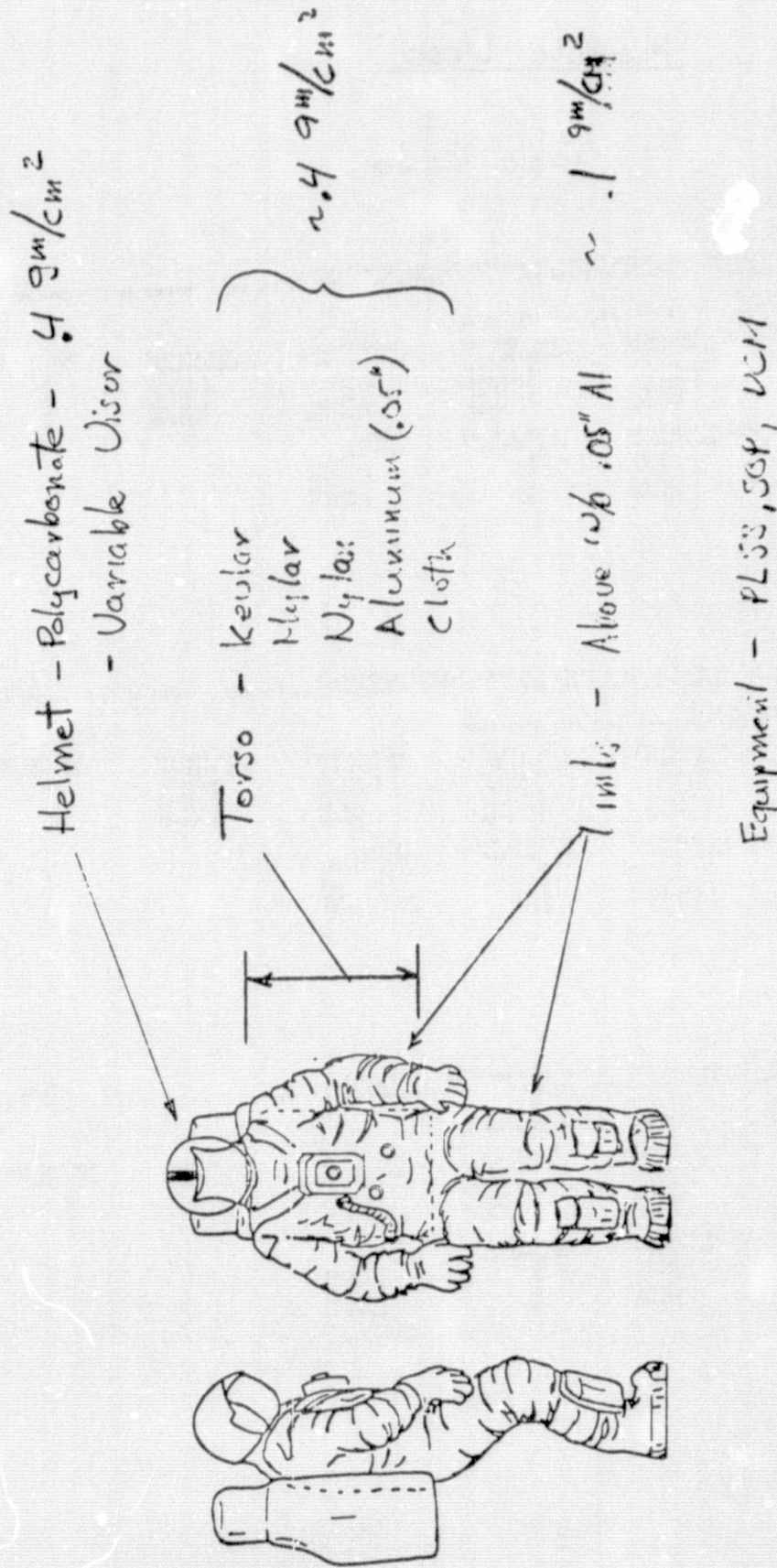


Figure 12-3. Shuttle EVA Suit

Module Dose

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Table 12-3a

SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 1 PAGE 8
SKIN - CAM 2.1
ORBIT - 500 M/270 NM/1 28.5 DEG INCLINATION

SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/50 CM	BREMSSTRAHLG 1.0 E/50 CM	REM AP6-AP7 1.0 P/50 CM	NOY 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
5.14E-01	2.82E-03	7.70E-05	2.11E-01	0.	0.
6.86E-01	1.15E-03	6.41E-05	1.90E-01	0.	0.
1.03E+00	8.18E-05	5.15E-05	1.64E-01	0.	0.
1.37E+00	9.03E-07	4.42E-05	1.53E-01	0.	0.
SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DOSE USE			
5.14E-01	0.	2.14E-01			
6.86E-01	0.	1.91E-01			
1.03E+00	0.	1.64E-01			
1.37E+00	0.	1.53E-01			

SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 1 PAGE 9
SKIN - CAM 2.1
ORBIT - 500 M/243 NM/1 28.5 DEG INCLINATION

SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/50 CM	BREMSSTRAHLG 1.0 E/50 CM	REM AP6-AP7 1.0 P/50 CM	NOY 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
5.14E-01	7.94E-03	2.29E-04	4.07E-01	0.	0.
6.86E-01	3.27E-03	1.90E-04	3.58E-01	0.	0.
1.03E+00	2.23E-04	1.52E-04	2.98E-01	0.	0.
1.37E+00	2.41E-06	1.50E-04	2.70E-01	0.	0.
SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DOSE USE			
5.14E-01	0.	4.16E-01			
6.86E-01	0.	3.61E-01			
1.03E+00	0.	2.98E-01			
1.37E+00	0.	2.70E-01			

SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 1 PAGE 10
SKIN - CAM 2.1
ORBIT - 500 M/270 NM/1 28.5 DEG INCLINATION

SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/50 CM	BREMSSTRAHLG 1.0 E/50 CM	REM AP6-AP7 1.0 P/50 CM	NOY 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
5.14E-01	1.62E-02	4.87E-04	7.17E-01	0.	0.
6.86E-01	6.52E-03	4.04E-04	6.13E-01	0.	0.
1.03E+00	4.49E-04	3.23E-04	4.98E-01	0.	0.
1.37E+00	4.84E-06	2.76E-04	4.40E-01	0.	0.
SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DOSE USE			
5.14E-01	0.	7.33E-01			
6.86E-01	0.	6.20E-01			
1.03E+00	0.	4.98E-01			
1.37E+00	0.	4.40E-01			

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Table 12-3b

H 05
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 1 PAGE H 05 11

1 1 1 SKIN - CAM 2.1
ORBIT 400 KM(216 NM) 48.0 DEG INCLINATION

N	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOY 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	4.51E-01	2.47E-03	5.61E-01	9.16E+01	2.94E+01
2	6.86E-01	2.69E-01	2.28E-03	2.93E-01	7.70E+01	2.48E+01
3	1.03E+00	1.08E-01	2.10E-03	2.25E-01	5.81E+01	1.90E+01
4	1.37E+00	4.60E-02	1.98E-03	1.96E-01	4.67E+01	1.58E+01
N	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL DOSE USE			
1	5.14E-01	3.94E+02	5.17E+02			
2	6.86E-01	3.32E+02	4.35E+02			
3	1.03E+00	2.56E+02	3.33E+02			
4	1.37E+00	2.12E+02	2.75E+02			

SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 1 PAGE H 06 12

SKIN - CAM 2.1
ORBIT 450 KM(243 NM) 48.0 DEG INCLINATION

N	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOY 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	5.46E-01	3.19E-03	5.74E-01	9.16E+01	2.94E+01
2	6.86E-01	3.24E-01	2.92E-03	4.55E-01	7.70E+01	2.48E+01
3	1.03E+00	1.29E-01	2.65E-03	3.43E-01	5.81E+01	1.90E+01
4	1.37E+00	5.47E-02	2.49E-03	2.92E-01	4.67E+01	1.58E+01
N	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL DOSE USE			
1	5.14E-01	3.94E+02	5.17E+02			
2	6.86E-01	3.32E+02	4.35E+02			
3	1.03E+00	2.56E+02	3.33E+02			
4	1.37E+00	2.12E+02	2.75E+02			

C 06
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 1 PAGE C 06 13

1 1 1 SKIN - CAM 2.1
ORBIT 500 KM(270 NM) 48.0 DEG INCLINATION

N	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOY 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	6.24E-01	3.93E-03	8.67E-01	9.16E+01	2.94E+01
2	6.86E-01	3.69E-01	3.56E-03	6.77E-01	7.70E+01	2.48E+01
3	1.03E+00	1.46E-01	3.21E-03	5.05E-01	5.81E+01	1.90E+01
4	1.37E+00	6.18E-02	2.99E-03	4.22E-01	4.67E+01	1.58E+01
N	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL DOSE USE			
1	5.14E-01	3.94E+02	5.17E+02			
2	6.86E-01	3.32E+02	4.35E+02			
3	1.03E+00	2.56E+02	3.33E+02			
4	1.37E+00	2.12E+02	2.75E+02			

Table 12-3c

1-ACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 1 PAGE 14
SKIN - CAM 2.1
ORBIT 450 KM (216 NM) 55.0 DEG INCLINATION

N	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/38 CM	BREMSSTRAHLE 1.0 E/38 CM	REM AP6-AP7 1.0 P/38 CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
1	5.14E-01	1.04E+00	4.99E-03	2.72E-01	3.56E+02	1.41E+02
2	6.86E-01	6.10E-01	4.71E-03	2.22E-01	2.85E+02	1.05E+02
3	1.03E+00	2.42E-01	4.43E-03	1.71E-01	1.99E+02	6.76E+01
4	1.37E+00	1.02E-01	4.25E-03	1.50E-01	1.52E+02	5.08E+01
N	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DONT USE			
1	5.14E-01	1.90E+03	2.39E+03			
2	6.86E-01	1.41E+03	1.80E+03			
3	1.03E+00	9.07E+02	1.17E+03			
4	1.37E+00	6.82E+02	8.85E+02			

1-ACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 1 PAGE 15
SKIN - CAM 2.1
ORBIT 450 KM (243 NM) 55.0 DEG INCLINATION

N	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/38 CM	BREMSSTRAHLE 1.0 E/38 CM	REM AP6-AP7 1.0 P/38 CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
1	5.14E-01	1.17E+00	5.79E-03	4.43E-01	3.56E+02	1.41E+02
2	6.86E-01	6.84E-01	5.45E-03	3.53E-01	2.85E+02	1.05E+02
3	1.03E+00	2.70E-01	5.10E-03	2.68E-01	1.99E+02	6.76E+01
4	1.37E+00	1.15E-01	4.88E-03	2.29E-01	1.52E+02	5.08E+01
N	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DONT USE			
1	5.14E-01	1.90E+03	2.39E+03			
2	6.86E-01	1.41E+03	1.80E+03			
3	1.03E+00	9.07E+02	1.18E+03			
4	1.37E+00	6.82E+02	8.85E+02			

1-ACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 1 PAGE 16
SKIN - CAM 2.1
ORBIT 500 KM (270 NM) 55.0 DEG INCLINATION

N	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/38 CM	BREMSSTRAHLE 1.0 E/38 CM	REM AP6-AP7 1.0 P/38 CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
1	5.14E-01	1.35E+00	6.86E-03	6.75E-01	3.56E+02	1.41E+02
2	6.86E-01	7.86E-01	6.42E-03	5.29E-01	2.85E+02	1.05E+02
3	1.03E+00	3.11E-01	5.98E-03	3.96E-01	1.99E+02	6.76E+01
4	1.37E+00	1.32E-01	5.70E-03	3.35E-01	1.52E+02	5.08E+01
N	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DONT USE			
1	5.14E-01	1.90E+03	2.39E+03			
2	6.86E-01	1.41E+03	1.80E+03			
3	1.03E+00	9.07E+02	1.18E+03			
4	1.37E+00	6.82E+02	8.85E+02			

Table 12-3d

1.0 CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
A MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 5

REF P4 - CAM 2.1
ORBIT 450 KM(241 NM) 28.5 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
1	5.14E-01	2.09E-16	2.05E-05	6.00E-02	0.	0.
2	6.86E-01	2.16E-20	2.01E-05	5.86E-02	0.	0.
3	1.03E+00	2.48E-28	1.93E-05	5.59E-02	0.	0.
4	1.37E+00	2.78E-30	1.87E-05	5.34E-02	0.	0.

	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DONT USE
1	5.14E-01	0.	6.01E-02
2	6.86E-01	0.	5.86E-02
3	1.03E+00	0.	5.59E-02
4	1.37E+00	0.	5.34E-02

SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 6

REF P4 - CAM 2.1
ORBIT 450 KM(241 NM) 28.5 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
1	5.14E-01	5.60E-16	5.93E-05	1.10E-01	0.	0.
2	6.86E-01	5.90E-20	5.81E-05	1.07E-01	0.	0.
3	1.03E+00	7.25E-28	5.60E-05	1.03E-01	0.	0.
4	1.37E+00	8.39E-30	5.40E-05	9.94E-02	0.	0.

	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DONT USE
1	5.14E-01	0.	1.10E-01
2	6.86E-01	0.	1.08E-01
3	1.03E+00	0.	1.03E-01
4	1.37E+00	0.	9.94E-02

B 08
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 7

1 1 1 RED BFO - CAM 2.1
ORBIT 500 KM(270 NM) 28.5 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
1	5.14E-01	1.13E-15	1.25E-04	1.79E-01	0.	0.
2	6.86E-01	1.22E-19	1.23E-04	1.76E-01	0.	0.
3	1.03E+00	1.52E-27	1.18E-04	1.70E-01	0.	0.
4	1.37E+00	1.79E-29	1.14E-04	1.64E-01	0.	0.

	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DONT USE
1	5.14E-01	0.	1.79E-01
2	6.86E-01	0.	1.76E-01
3	1.03E+00	0.	1.70E-01
4	1.37E+00	0.	1.64E-01

Table 12-3e

C 08
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 8

1 1 1 RED BFO - CAM 2.1 ORBIT 400 KM(216 NM) 48.0 DEG INCLINATION							
N 1 2 3 4	SHIELD THICKNESS	REM AES-AE7	BREMSSTRAHLG	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	1.0 E/SQ CM	1.0 E/SQ CM	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	1.38E-04	1.39E-03	6.19E-02	4.52E+00	3.54E+00	
	6.86E-01	7.35E-05	1.37E-03	6.03E-02	4.25E+00	3.44E+00	
N 1 2 3 4	SHIELD THICKNESS	STAT SCR MAX	TOTAL	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	99CL 6MO REM	DONT USE	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	4.76E+01	5.57E+01	6.19E-02	4.52E+00	3.54E+00	
	6.86E-01	4.61E+01	5.39E+01	6.03E-02	4.25E+00	3.44E+00	
N 1 2 3 4	SHIELD THICKNESS	STAT SCR MAX	TOTAL	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	99CL 6MO REM	DONT USE	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	4.76E+01	5.57E+01	6.19E-02	4.52E+00	3.54E+00	
	6.86E-01	4.61E+01	5.39E+01	6.03E-02	4.25E+00	3.44E+00	
N 1 2 3 4	SHIELD THICKNESS	STAT SCR MAX	TOTAL	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	99CL 6MO REM	DONT USE	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	4.76E+01	5.57E+01	6.19E-02	4.52E+00	3.54E+00	
	6.86E-01	4.61E+01	5.39E+01	6.03E-02	4.25E+00	3.44E+00	

J 08
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 9

1 1 1 RED BFO - CAM 2.1 ORBIT 450 KM(243 NM) 48.0 DEG INCLINATION							
N 1 2 3 4	SHIELD THICKNESS	REM AES-AE7	BREMSSTRAHLG	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	1.0 E/SQ CM	1.0 E/SQ CM	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	1.83E-04	1.71E-03	9.55E-02	4.52E+00	3.54E+00	
	6.86E-01	8.70E-05	1.69E-03	9.32E-02	4.25E+00	3.44E+00	
N 1 2 3 4	SHIELD THICKNESS	STAT SCR MAX	TOTAL	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	99CL 6MO REM	DONT USE	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	4.76E+01	5.57E+01	9.55E-02	4.52E+00	3.54E+00	
	6.86E-01	4.61E+01	5.39E+01	9.32E-02	4.25E+00	3.44E+00	
N 1 2 3 4	SHIELD THICKNESS	STAT SCR MAX	TOTAL	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	99CL 6MO REM	DONT USE	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	4.76E+01	5.57E+01	9.55E-02	4.52E+00	3.54E+00	
	6.86E-01	4.61E+01	5.39E+01	9.32E-02	4.25E+00	3.44E+00	

E 08
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 10

1 1 1 RED BFO - CAM 2.1 ORBIT 500 KM(270 NM) 48.0 DEG INCLINATION							
N 1 2 3 4	SHIELD THICKNESS	REM AES-AE7	BREMSSTRAHLG	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	1.0 E/SQ CM	1.0 E/SQ CM	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	1.83E-04	2.00E-03	1.41E-01	4.52E+00	3.54E+00	
	6.86E-01	9.79E-05	1.92E-03	1.38E-01	4.25E+00	3.44E+00	
N 1 2 3 4	SHIELD THICKNESS	STAT SCR MAX	TOTAL	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	99CL 6MO REM	DONT USE	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	4.76E+01	5.58E+01	1.41E-01	4.52E+00	3.54E+00	
	6.86E-01	4.61E+01	5.40E+01	1.38E-01	4.25E+00	3.44E+00	
N 1 2 3 4	SHIELD THICKNESS	STAT SCR MAX	TOTAL	REM AP6-AP7	NOV 1960 SCR	STAT SCR MIN	
	GRAMS/SQCM	99CL 6MO REM	DONT USE	1.0 P/SQ CM	(REM/EVENT)	99CL 6MO REM	
	5.14E-01	4.76E+01	5.58E+01	1.41E-01	4.52E+00	3.54E+00	
	6.86E-01	4.61E+01	5.40E+01	1.38E-01	4.25E+00	3.44E+00	

Table 12-3F

1-2, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS EYL WALL MODULE IS 15 M LONG AND 4.4 M DIA

*CASE 2, PAGE 11

1 1 1 RED BFO - CAN 2.1 ORBIT 410 KM(216 NM) 55.0 DEG INCLINATION						
	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/50 CM	BREMSSTRAHLG 1.0 E/50 CM	REM APG-AP7 1.0 P/50 CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	3.04E-04	3.12E-03	4.79E-02	1.08E+01	7.49E+00
2	6.86E-01	1.62E-04	3.08E-03	4.66E-02	1.00E+01	7.21E+00
3	1.03E+00	4.03E-05	3.02E-03	4.42E-02	8.76E+00	6.72E+00
4	1.37E+00	7.92E-07	2.96E-03	4.21E-02	7.73E+00	6.29E+00
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL BOMB USE			
1	5.14E-01	1.01E+02	1.19E+02			
2	6.86E-01	9.68E+01	1.14E+02			
3	1.03E+00	9.02E+01	1.06E+02			
4	1.37E+00	8.44E+01	9.85E+01			

3-2, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS EYL WALL MODULE IS 15 M LONG AND 4.4 M DIA

*CASE 2, PAGE 12

1 1 1 RED BFO - CAN 2.1 ORBIT 450 KM(243 NM) 55.0 DEG INCLINATION						
	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/50 CM	BREMSSTRAHLG 1.0 E/50 CM	REM APG-AP7 1.0 P/50 CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	3.40E-04	3.34E-03	7.69E-02	1.11E+01	7.49E+00
2	6.86E-01	1.81E-04	3.30E-03	7.51E-02	1.01E+01	7.21E+00
3	1.03E+00	4.50E-05	3.43E-03	7.19E-02	8.76E+00	6.72E+00
4	1.37E+00	8.84E-07	3.36E-03	6.89E-02	7.73E+00	6.29E+00
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL BOMB USE			
1	5.14E-01	1.01E+02	1.19E+02			
2	6.86E-01	9.68E+01	1.14E+02			
3	1.03E+00	9.02E+01	1.06E+02			
4	1.37E+00	8.44E+01	9.85E+01			

4-2, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS EYL WALL MODULE IS 15 M LONG AND 4.4 M DIA

*CASE 2, PAGE 13

1 1 1 RED BFO - CAN 2.1 ORBIT 500 KM(270 NM) 55.0 DEG INCLINATION						
	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/50 CM	BREMSSTRAHLG 1.0 E/50 CM	REM APG-AP7 1.0 P/50 CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	3.91E-04	4.10E-03	1.12E-01	1.08E+01	7.49E+00
2	6.86E-01	2.08E-04	4.03E-03	1.12E-01	1.00E+01	7.21E+00
3	1.03E+00	5.17E-05	3.97E-03	1.04E-01	8.76E+00	6.72E+00
4	1.37E+00	1.01E-06	3.89E-03	1.04E-01	7.73E+00	6.29E+00
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL BOMB USE			
1	5.14E-01	1.01E+02	1.19E+02			
2	6.86E-01	9.68E+01	1.14E+02			
3	1.03E+00	9.02E+01	1.06E+02			
4	1.37E+00	8.44E+01	9.86E+01			

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Table 12-3g

E 02
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 35

4 4 4 LENS OF EYE - CAM 2.1
ORBIT 400 KM(216 NM) 28.5 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SR CM	BREMSSTRAHLG 1.0 E/SR CM	REM AP6-AP7 1.0 P/SR CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	2.51E-04	5.08E-05	1.70E-01	0.	0.
2	6.86E-01	3.57E-05	5.21E-05	1.70E-01	0.	0.
3	1.03E+00	3.59E-07	4.49E-05	1.60E-01	0.	0.
4	1.37E+00	3.95E-09	3.99E-05	1.54E-01	0.	0.
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL DONT USE			
1	5.14E-01	0.	1.70E-01			
2	6.86E-01	0.	1.70E-01			
3	1.03E+00	0.	1.60E-01			
4	1.37E+00	0.	1.54E-01			

F 02
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 36

4 4 4 LENS OF EYE - CAM 2.1
ORBIT 450 KM(243 NM) 28.5 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SR CM	BREMSSTRAHLG 1.0 E/SR CM	REM AP6-AP7 1.0 P/SR CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	6.87E-04	1.68E-04	3.16E-01	0.	0.
2	6.86E-01	9.14E-05	1.53E-04	3.00E-01	0.	0.
3	1.03E+00	9.57E-07	1.32E-04	2.75E-01	0.	0.
4	1.37E+00	1.03E-08	1.17E-04	2.58E-01	0.	0.
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL DONT USE			
1	5.14E-01	0.	3.16E-01			
2	6.86E-01	0.	3.00E-01			
3	1.03E+00	0.	2.75E-01			
4	1.37E+00	0.	2.58E-01			

G 02
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 37

4 4 4 LENS OF EYE - CAM 2.1
ORBIT 500 KM(270 NM) 28.5 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SR CM	BREMSSTRAHLG 1.0 E/SR CM	REM AP6-AP7 1.0 P/SR CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	1.39E-03	3.56E-04	5.22E-01	0.	0.
2	6.86E-01	1.84E-04	3.26E-04	4.90E-01	0.	0.
3	1.03E+00	1.92E-06	2.80E-04	4.38E-01	0.	0.
4	1.37E+00	2.06E-08	2.48E-04	4.03E-01	0.	0.
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL DONT USE			
1	5.14E-01	0.	5.22E-01			
2	6.86E-01	0.	4.90E-01			
3	1.03E+00	0.	4.39E-01			
4	1.37E+00	0.	4.03E-01			

Table 12-3h

M 02
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2 PAGE 38

4 4 4 LENS OF EYE - CAM 2.1
ORBIT 400 KM(216 NM) 48.0 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOY 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	1.36E-01	2.25E-03	2.37E-01	5.98E-01	1.97E+01
2	6.86E-01	8.62E-02	2.18E-03	2.21E-01	5.33E-01	1.79E+01
3	1.03E+00	3.55E-02	2.07E-03	1.96E-01	4.28E-01	1.50E+01
4	1.37E+00	1.44E-02	1.97E-03	1.80E-01	3.49E-01	1.29E+01
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL DONT USE			
1	5.14E-01	2.65E+02	3.45E+02			
2	6.86E-01	2.40E+02	3.12E+02			
3	1.03E+00	2.01E+02	2.59E+02			
4	1.37E+00	1.73E+02	2.21E+02			

B 03
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2 PAGE 39

4 4 4 LENS OF EYE - CAM 2.1
ORBIT 450 KM(243 NM) 48.0 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOY 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	1.62E-01	2.86E-03	3.59E-01	5.98E-01	1.97E+01
2	6.86E-01	1.03E-01	2.76E-03	3.30E-01	5.33E-01	1.79E+01
3	1.03E+00	4.21E-02	2.59E-03	2.86E-01	4.28E-01	1.50E+01
4	1.37E+00	1.70E-02	2.47E-03	2.58E-01	3.49E-01	1.29E+01
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL DONT USE			
1	5.14E-01	2.65E+02	3.45E+02			
2	6.86E-01	2.40E+02	3.12E+02			
3	1.03E+00	2.01E+02	2.59E+02			
4	1.37E+00	1.73E+02	2.21E+02			

C 03
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2 PAGE 40

4 4 4 LENS OF EYE - CAM 2.1
ORBIT 500 KM(270 NM) 48.0 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOY 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL 6MO REM
1	5.14E-01	1.84E-01	3.46E-03	5.24E-01	5.98E-01	1.97E+01
2	6.86E-01	1.16E-01	3.32E-03	4.77E-01	5.33E-01	1.79E+01
3	1.03E+00	4.75E-02	3.11E-03	4.07E-01	4.28E-01	1.50E+01
4	1.37E+00	1.91E-02	2.94E-03	3.61E-01	3.49E-01	1.29E+01
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL 6MO REM	TOTAL DONT USE			
1	5.14E-01	2.65E+02	3.45E+02			
2	6.86E-01	2.40E+02	3.12E+02			
3	1.03E+00	2.01E+02	2.59E+02			
4	1.37E+00	1.73E+02	2.21E+02			

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Table 12-3i

D 03
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 47

4 4 4 LENS OF EYE - CAM 2.1
ORBIT 400 KM(216 NM) 55.0 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
1	5.14E-01	3.04E-01	4.74E-03	1.80E-01	2.03E+02	6.87E+01
2	6.86E-01	1.92E-01	4.63E-03	1.69E-01	1.76E+02	5.91E+01
3	1.03E+00	7.89E-02	4.45E-03	1.50E-01	1.35E+02	4.51E+01
4	1.37E+00	3.18E-02	4.30E-03	1.38E-01	1.05E+02	3.60E+01
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DONT USE			
1	5.14E-01	9.22E+02	1.19E+03			
2	6.86E-01	7.94E+02	1.03E+03			
3	1.03E+00	6.05E+02	7.85E+02			
4	1.37E+00	4.83E+02	6.24E+02			

E 03
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 42

4 4 4 LENS OF EYE - CAM 2.1
ORBIT 450 KM(243 NM) 55.0 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
1	5.14E-01	3.40E-01	5.46E-03	2.80E-01	2.03E+02	6.87E+01
2	6.86E-01	2.15E-01	5.33E-03	2.58E-01	1.76E+02	5.91E+01
3	1.03E+00	8.82E-02	5.10E-03	2.25E-01	1.35E+02	4.51E+01
4	1.37E+00	3.56E-02	4.92E-03	2.03E-01	1.05E+02	3.60E+01
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DONT USE			
1	5.14E-01	9.22E+02	1.19E+03			
2	6.86E-01	7.94E+02	1.03E+03			
3	1.03E+00	6.05E+02	7.85E+02			
4	1.37E+00	4.83E+02	6.24E+02			

F 03
SIGMA, SPACE RADIATION DOSE CALCULATIONS IN QUADRIC SURFACE GEOMETRIES
SPACE STATION MODULE DOSE PER DAY SHIELD THICKNESS IS CYL WALL MODULE IS 15 M LONG AND 4.4 M DIA *CASE 2, PAGE 43

4 4 4 LENS OF EYE - CAM 2.1
ORBIT 500 KM(270 NM) 55.0 DEG INCLINATION

	SHIELD THICKNESS GRAMS/SQCM	REM AES-AE7 1.0 E/SQ CM	BREMSSTRAHLG 1.0 E/SQ CM	REM AP6-AP7 1.0 P/SQ CM	NOV 1960 SCR (REM/EVENT)	STAT SCR MIN 99CL GMD REM
1	5.14E-01	3.92E-01	6.41E-03	4.12E-01	2.03E+02	6.87E+01
2	6.86E-01	2.48E-01	6.24E-03	3.76E-01	1.76E+02	5.91E+01
3	1.03E+00	1.02E-01	5.96E-03	3.22E-01	1.35E+02	4.51E+01
4	1.37E+00	4.10E-02	5.73E-03	2.86E-01	1.05E+02	3.60E+01
	SHIELD THICKNESS GRAMS/SQCM	STAT SCR MAX 99CL GMD REM	TOTAL DONT USE			
1	5.14E-01	9.22E+02	1.19E+03			
2	6.86E-01	7.94E+02	1.03E+03			
3	1.03E+00	6.05E+02	7.85E+02			
4	1.37E+00	4.83E+02	6.24E+02			

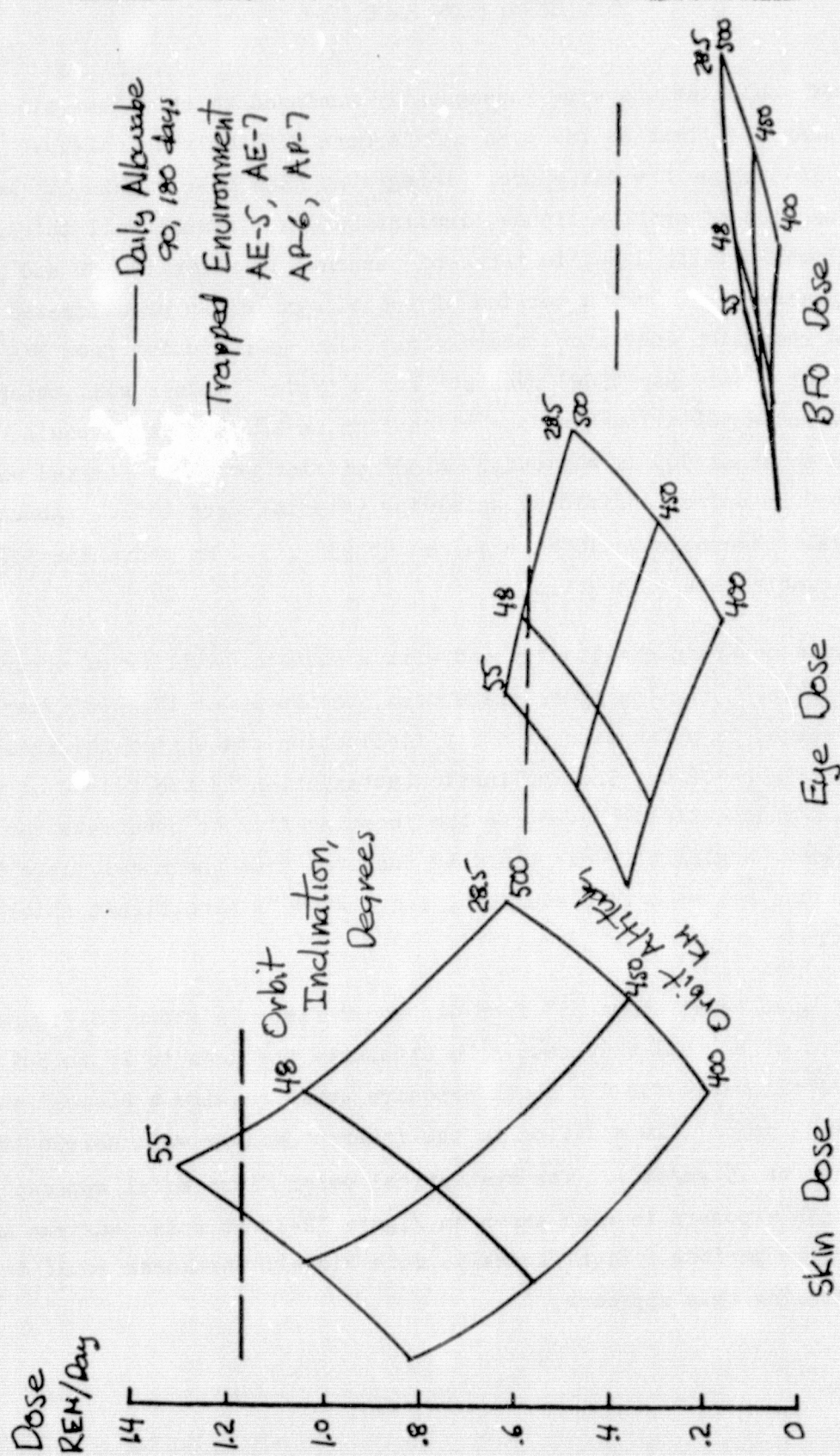


Figure 12-4. Module Radiation Dose — 0.1 In. Wall Thickness

The dose calculations were subsequently confined to skin dose since it is as near the limit as the eyes and is more difficult to shield. Figure 12-5 shows the daily orbit integrated skin dose within the module as a function of orbit altitude, inclination, and module wall thickness. The allowable daily dose, in terms of percent, is shown for 30 and 90 to 180 day missions. Only a portion of the allowable can be allocated to the trapped radiation dose since there might also be radiation from EVA and from Solar Cosmic Ray (SCR) events. For a typical module wall thickness of 0.1 in. and a 50% allowable dose allocation, a 28.5° mission would be limited to about 500 km altitude. A 55° mission would be limited well below 400 km unless additional shielding material were added. About 0.18 in. total wall thickness would be required at 55° x 500 km orbit for 50% of the allowable dose allocation.

SCR events occur intermittently and with a size-probability of occurrence relationship. For the model flare used, the November 12, 1960 flare, the dose received is a function of orbit inclination and shield thickness as shown in Figure 12-6. The inclination attenuation is the result of the earth's magnetic field preventing the incoming flux to penetrate the lower latitudes. Negligible dose would be received from the model flare for inclinations below 40°. Above that, i.e., at 55°, significant shield would be required.

The skin dose received at 55° from the model flare is shown in Figure 12-7 as a function of biowell thickness. To attenuate the dose to 25 to 50% of the total allowable for a 6 month exposure would require a biowell thickness of 1.5 to 3 gm/cm², in addition to the inherent module wall shield (here assumed to be .5 gm/cm²). The statistical solar flare model approach for a six-month exposure is also shown on Figure 12-7 for solar maximum and minimum time periods. Significantly, more Biowell thickness would be required using this approach.

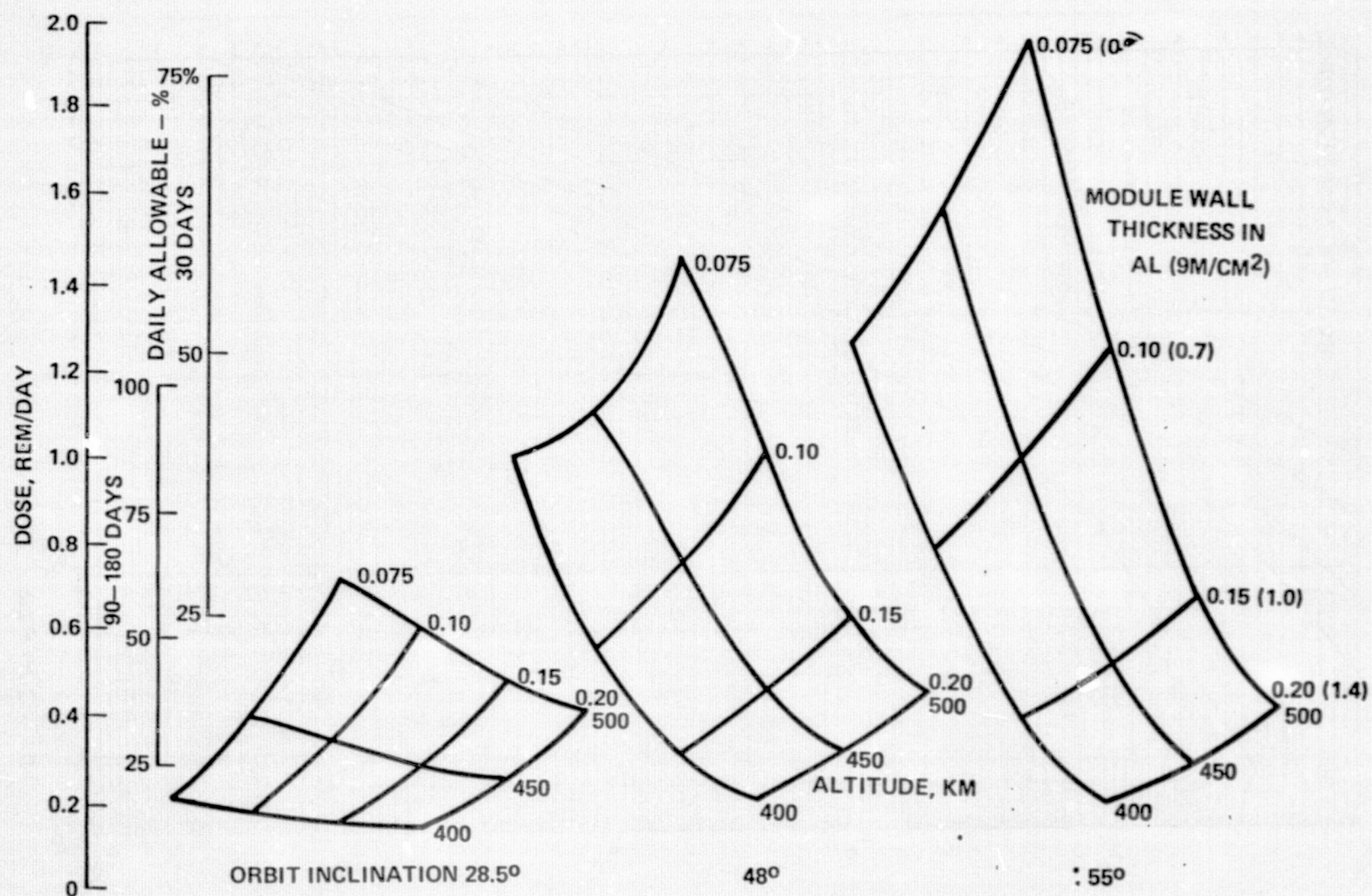


Figure 12-5. Module Skin Dose - Trapped Radiation

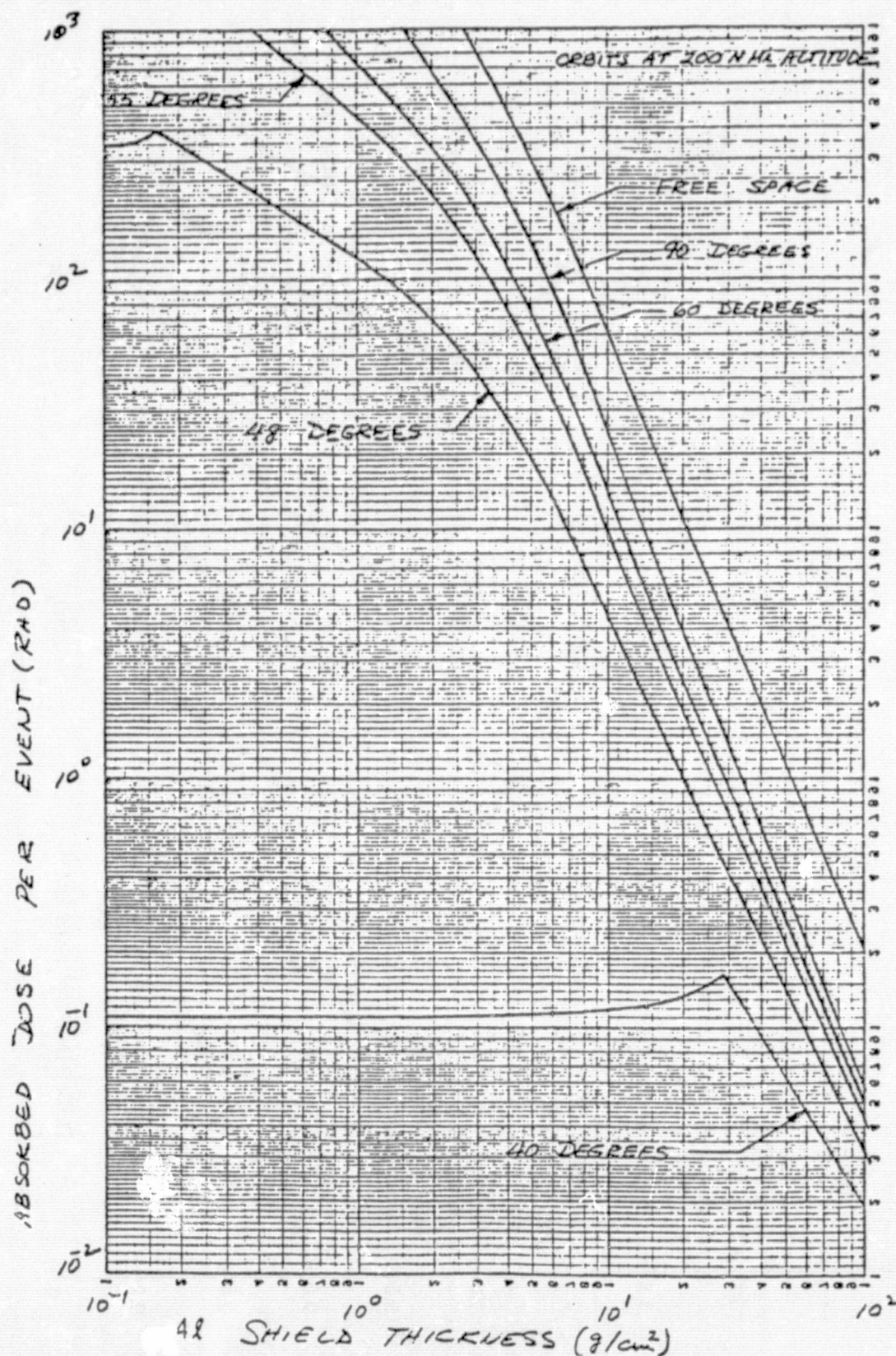
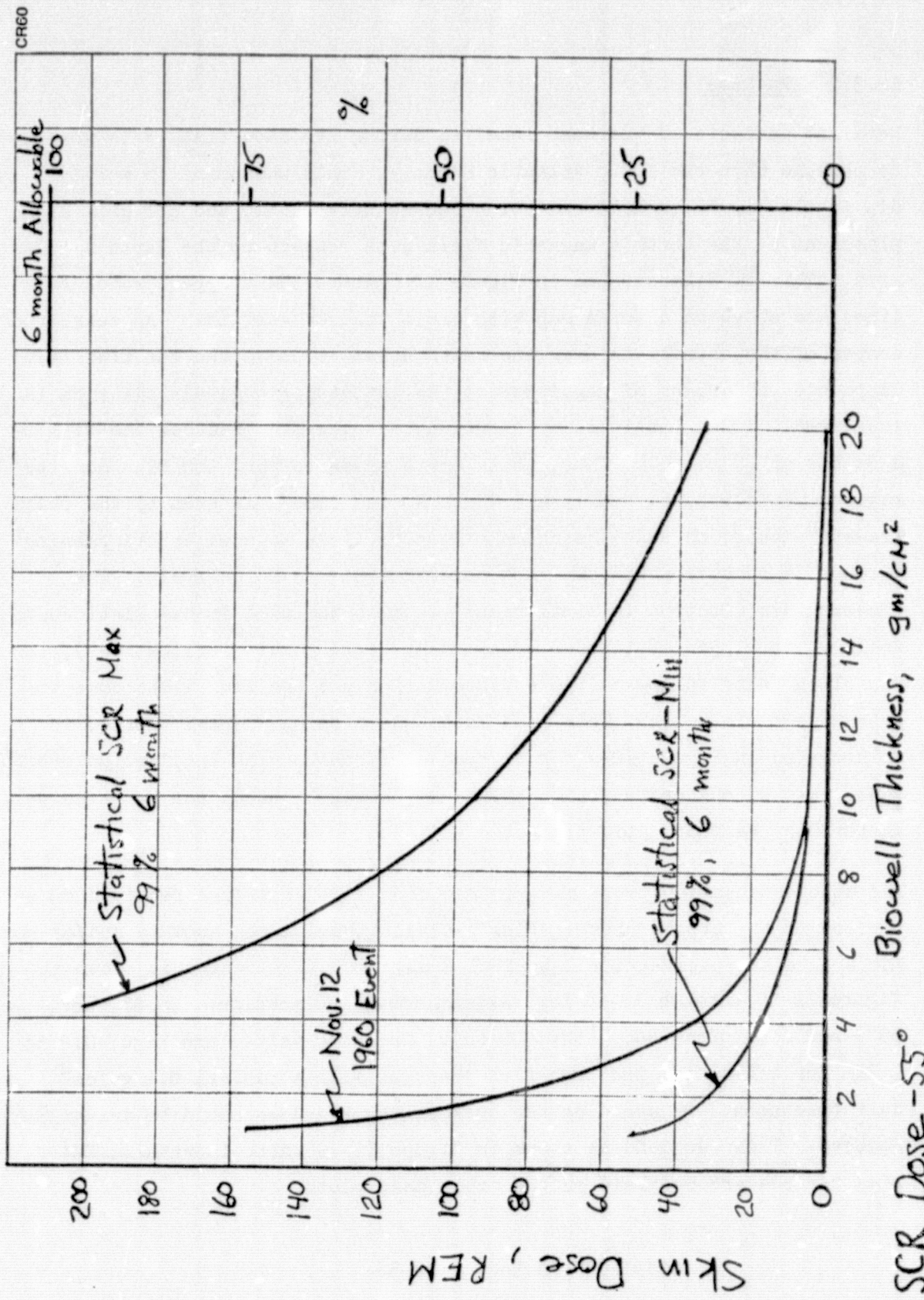


Figure 12-6. Proton Dose from Nov 12, 1960 Solar Cosmic Ray Event

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OF POOR QUALITY



12.3.2 EVA Dose

Detailed analysis of the dose received during EVA show that it is governed by passage thru the South Atlantic anomaly. This phenomena is merely a low dip in the Van Allen Belt radiation caused by the tilt and off axis displacement of the earth's magnetic field with respect to the earth's polar axis. This is demonstrated in Figure 12-8 where the trapped proton flux lines are shown on a world map with orbit traces overlaid. As seen, the center of the anomaly is over the South Atlantic Ocean and the flux level is reduced by orders of magnitude as the latitude/longitude distance is increased. A 30° latitude or longitude displacement reduces the flux by a factor of 20 to 100. Thus, those orbit passages thru the high density regions would produce nearly all the absorbed dose. As seen by the orbit traces, passage through the anomaly is periodic - several orbits passing through it in succession and then several avoiding it entirely. The electron distribution is similar and was also included in calculations to follow. At higher latitudes (reached by higher orbit inclinations), the Van Allen belts follow a dipole distribution and the flux lines come together near the poles. This high latitude dipping, termed horns, is illustrated in Figure 12-9 for trapped electrons. High inclination orbits would thus encounter radiation pulses by passage through the horns as well as through the South Atlantic anomaly.

The dose calculation programs were modified to output dose received on each portion of the orbit, thus showing passages through the anomaly and/or horns. The skin dose received for a typical 3 days worth of orbits is shown in Figure 12-10 through 12-18 for various orbit inclinations (28.5° , 48° , 55°) and altitudes (400, 450, 500 km). The dose values are high, 0.2 to 3 REM per orbit, for the high flux passages with a nominal 0.1 gm/cm^2 suit thickness. By avoiding the spikes, the EVA dose could be dramatically reduced. This was done as shown in Figure 12-19 where increasing EVA time periods are overlaid on the orbit dose plots.

CONTINUED ON PAGE 12-35

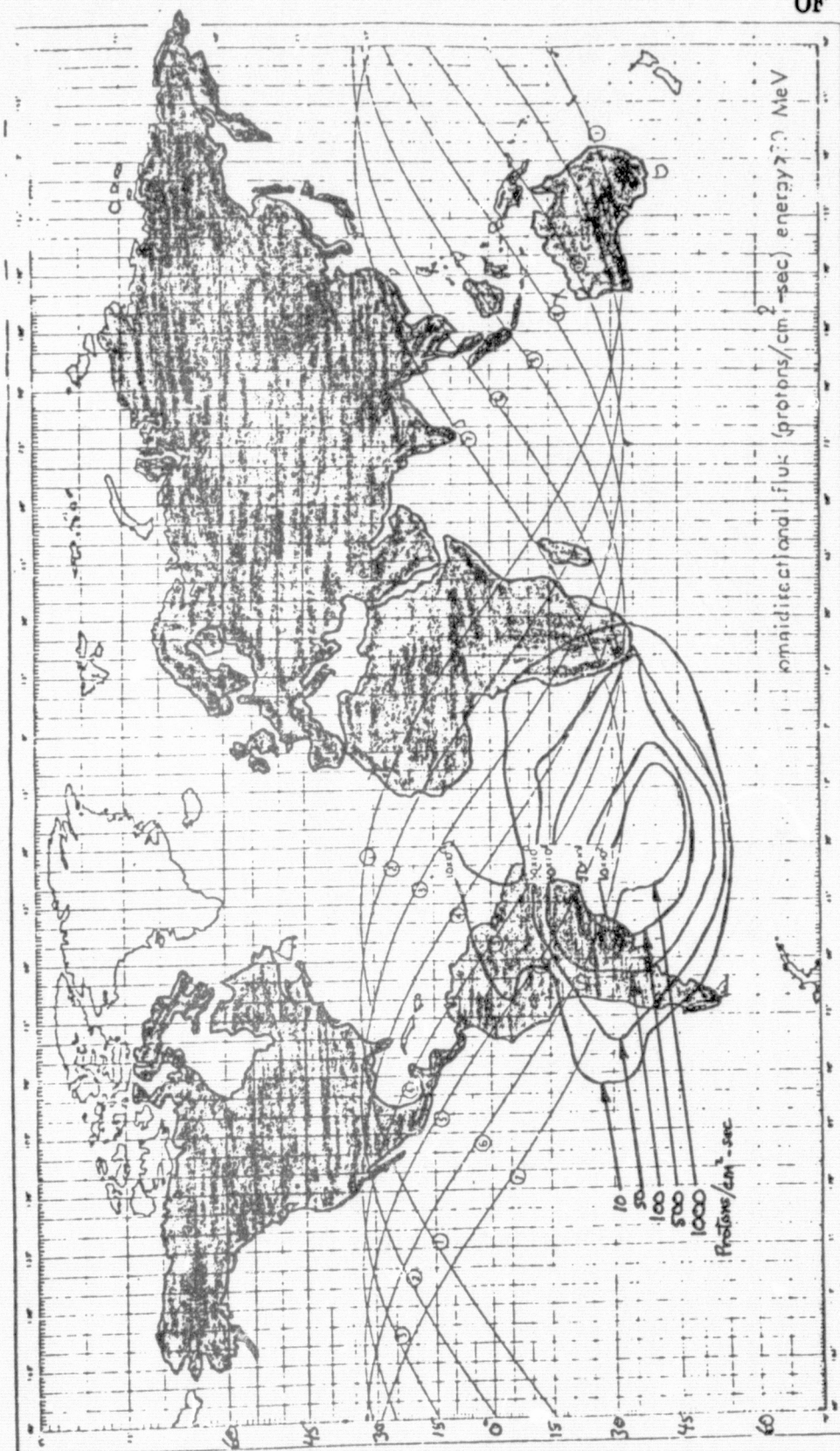


Figure 12-8. Proton Flux Density (300 km)

Electron Flux Density - ($E \geq 0.5$ MEU)

CR60

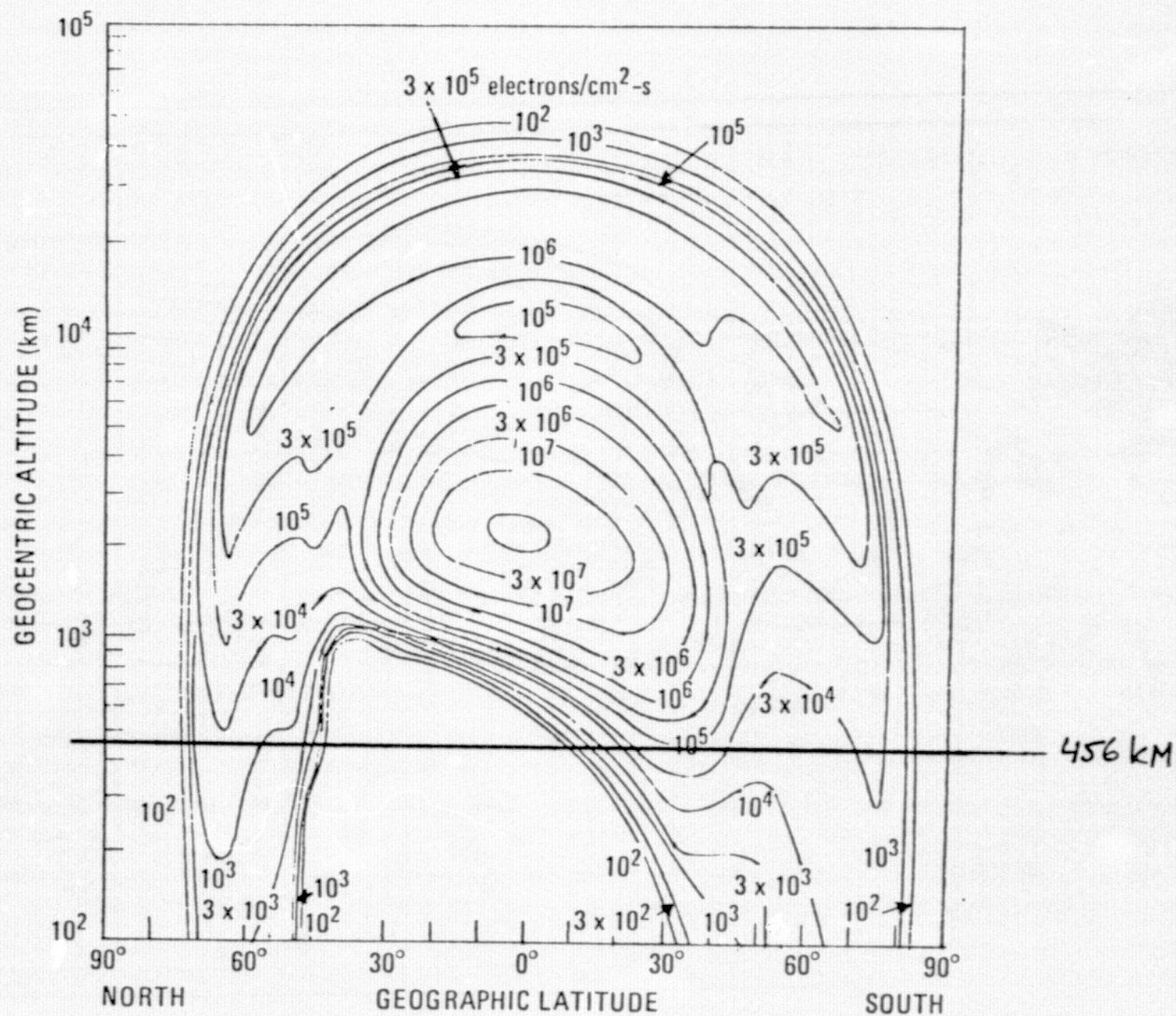


Figure 12-9. Electron Flux Density - ($E \geq 0.5$ MEU)

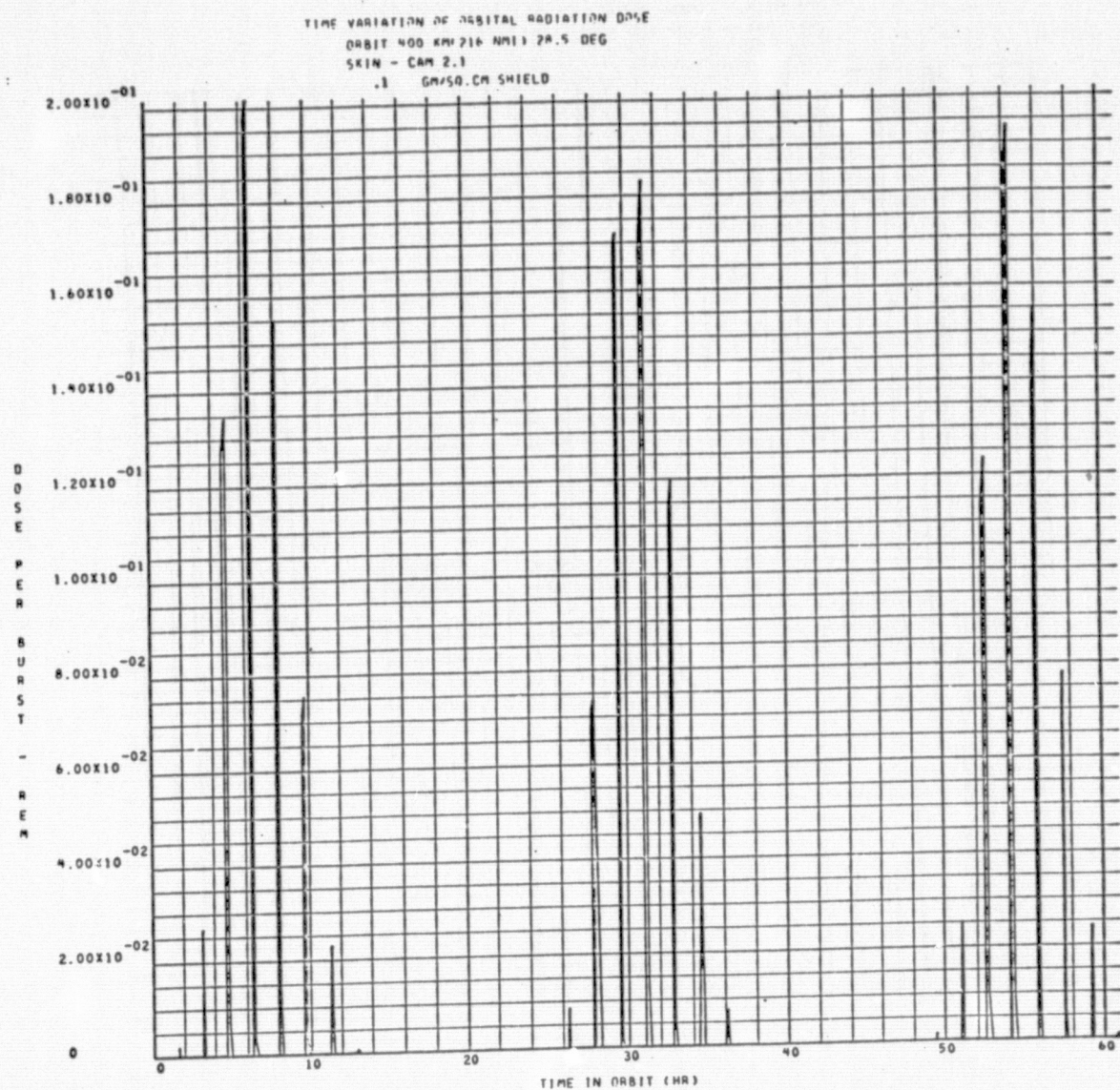


Figure 12-10. Time Variation of Orbital Radiation Dose Orbit 400 km (216 nmi) 28.5 Degree Skin — Cam 2.1

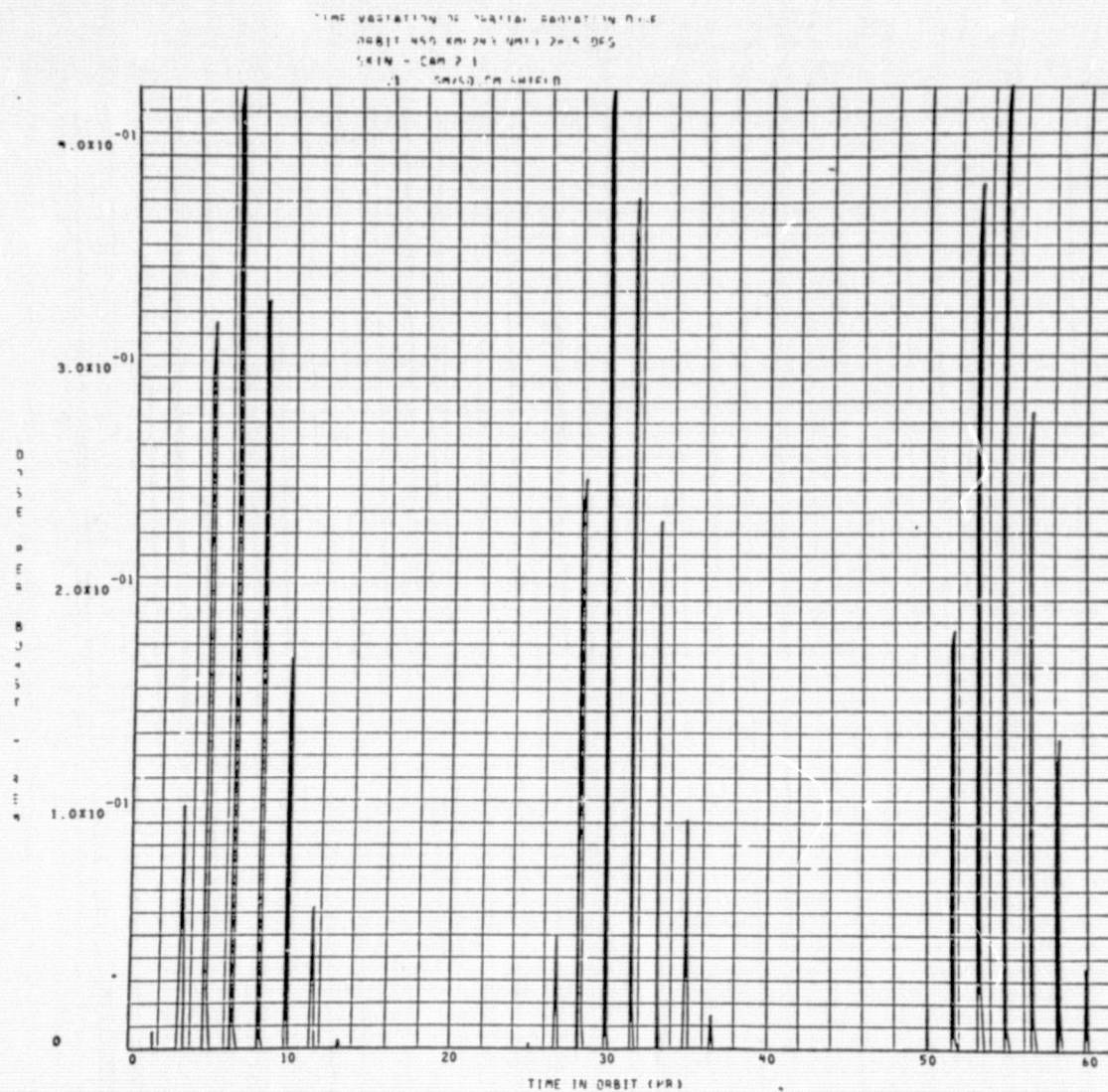


Figure 12-11. Time Variation of Orbital Radiation Dose Orbit 450 km (243 nmi) 28.5 Degree Skin — Cam 2.1

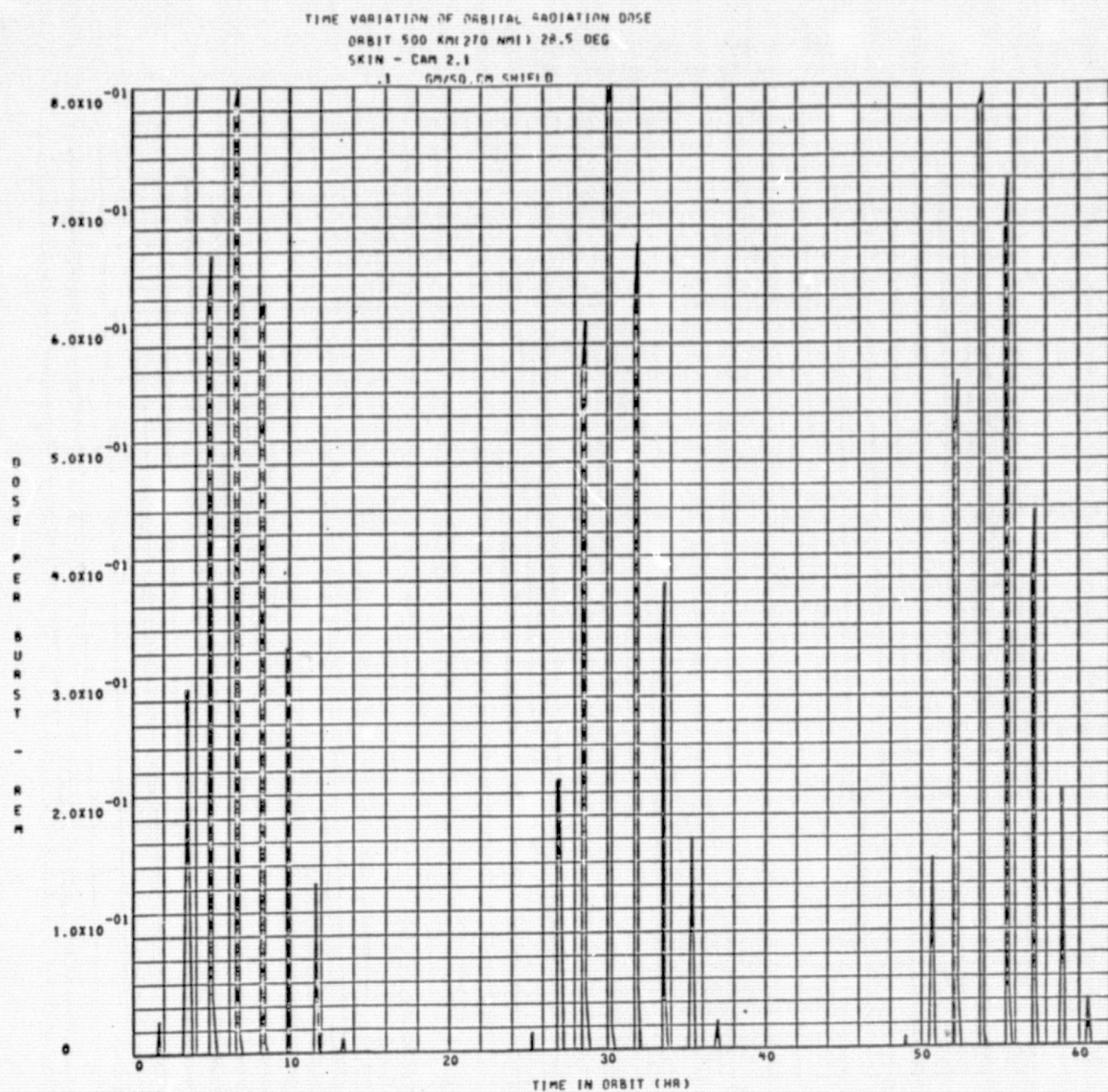


Figure 12-12. Time Variation of Orbital Radiation Dose Orbit 500 km (270 nmi) 28.5 Deg Skin - Cam 2.1

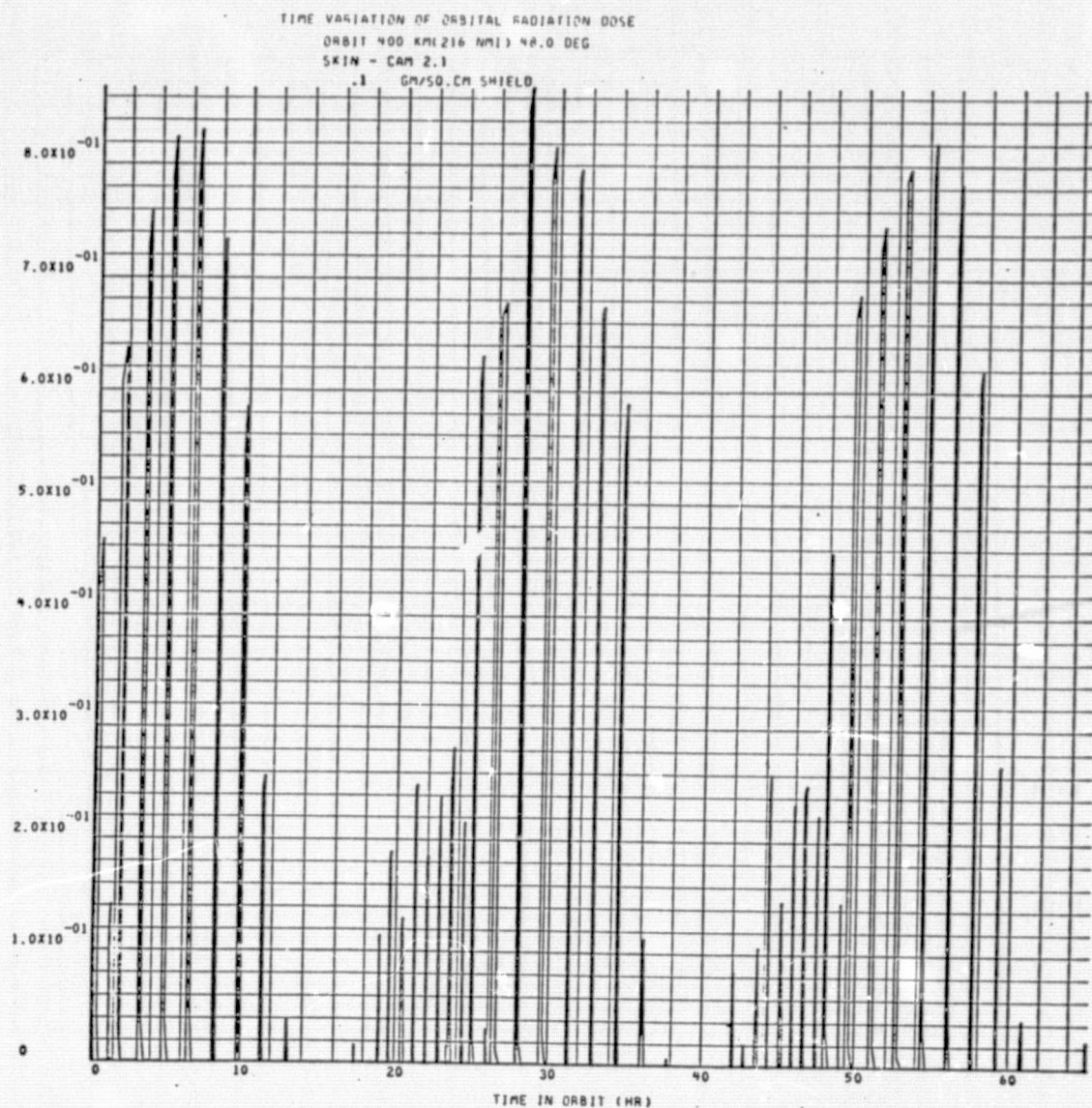


Figure 12-13. Time Variation of Orbital Radiation Dose Orbit 400 km (216 nmi) 48.0 Deg Skin - Cam 2.1

TIME VARIATION OF ORBITAL RADIATION DOSE
 ORBIT 450 KM (243 NMI) 48.0 DEG
 SKIN - CAM 2.1
 .1 GR/50 CM SHIELD

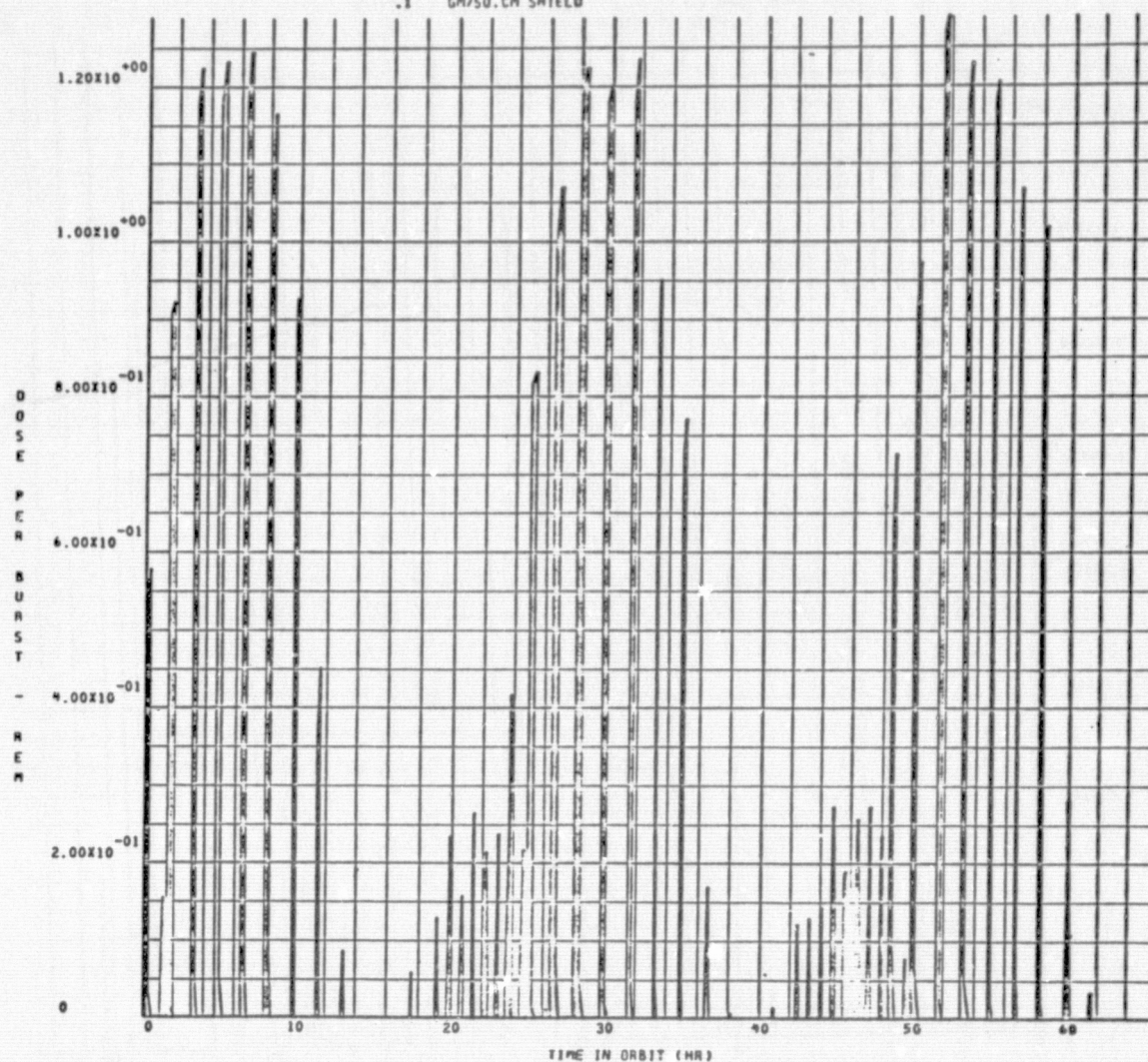


Figure 12-14. Time Variation of Orbital Radiation Dose Orbit 450 km (243 nmi) 48.0 Degree Skin - Cam 2.1

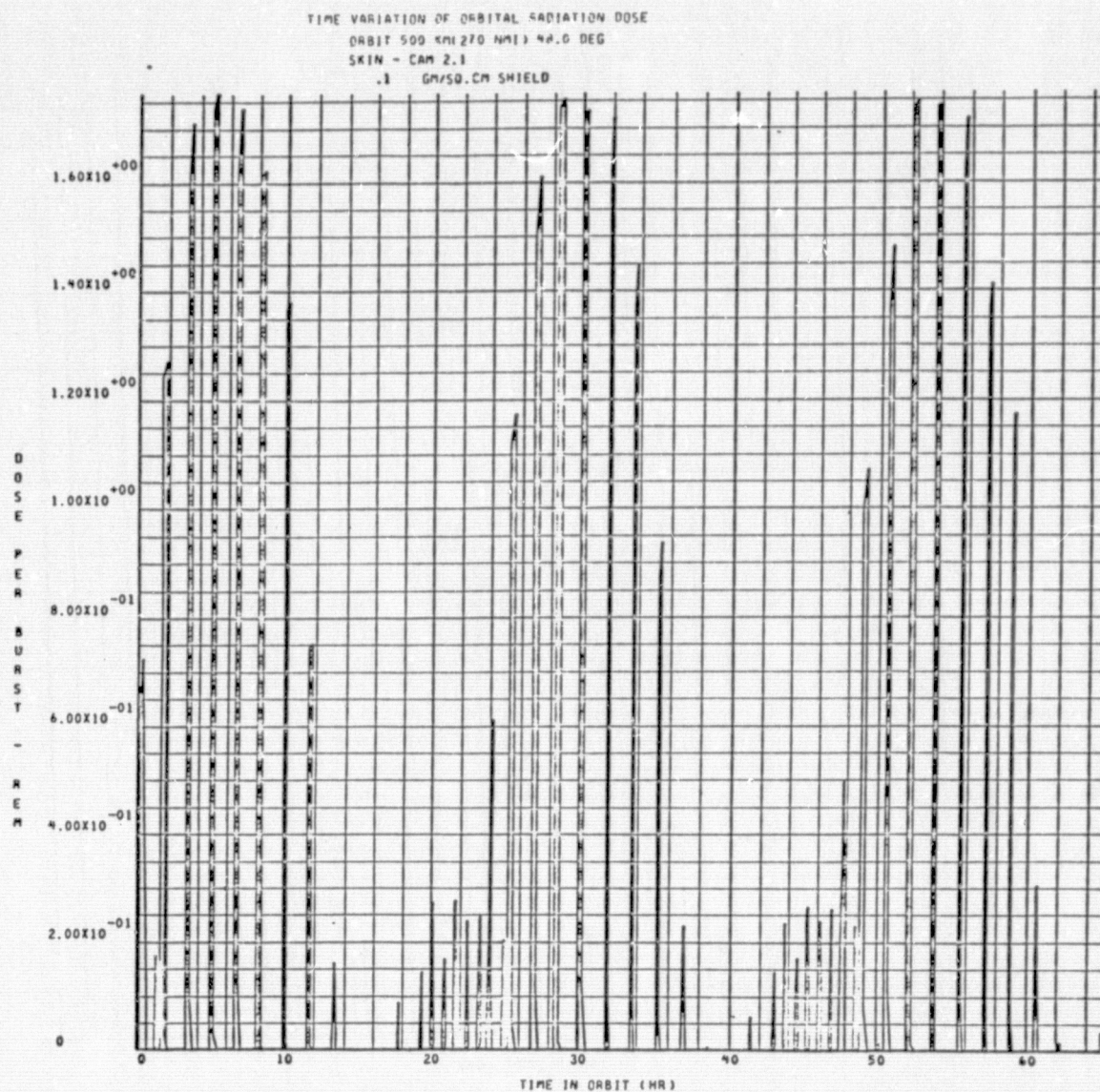


Figure 12-15. Time Variation of Orbital Radiation Dose Orbit 500 km (270 nmi) 48.0 Deg Skin - Cam 2.1

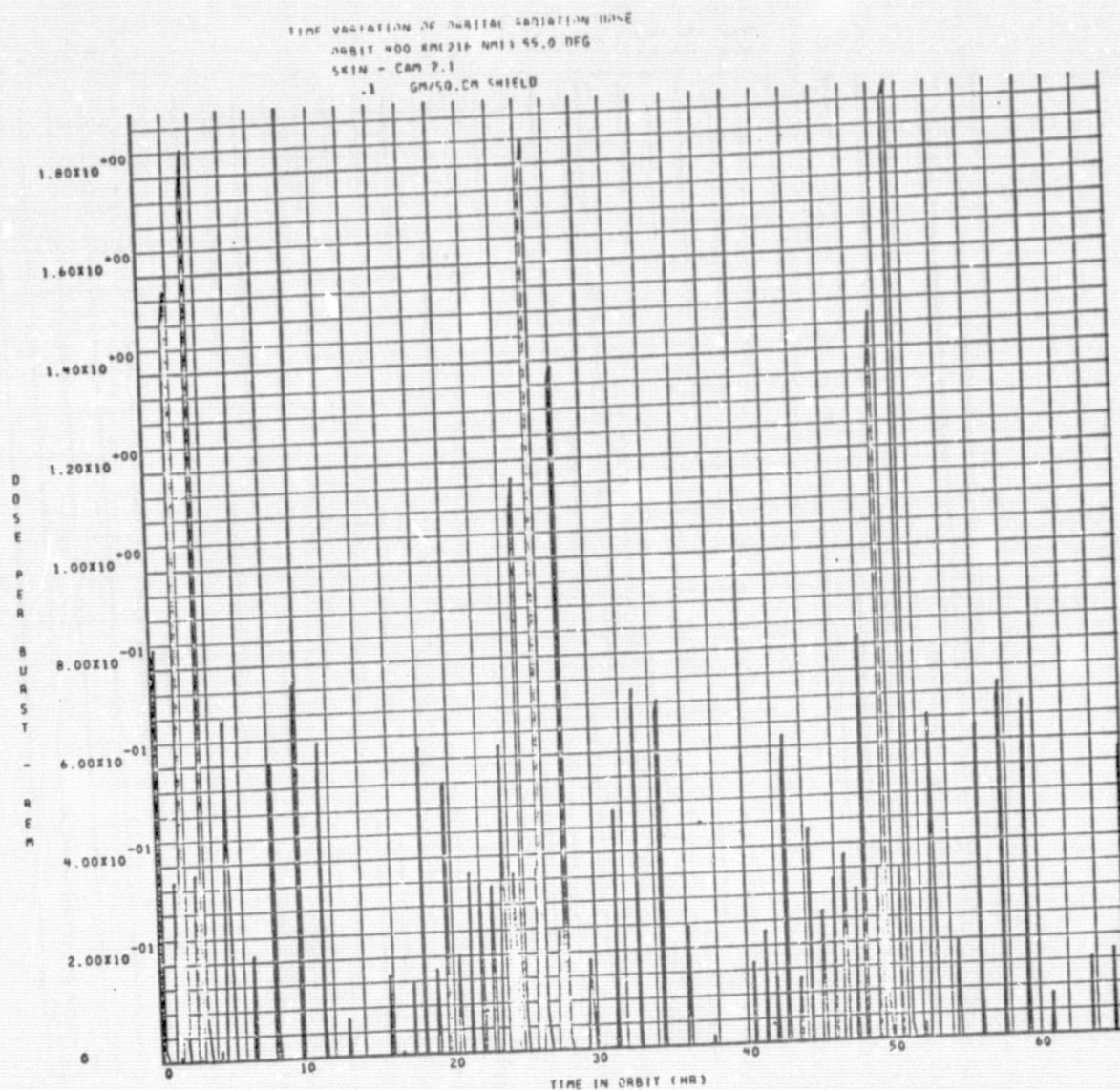


Figure 12-16. Time Variation of Orbital Radiation Dose Orbit 400 km (216 nmi) 55.0 Deg Skin — Cam 2.1

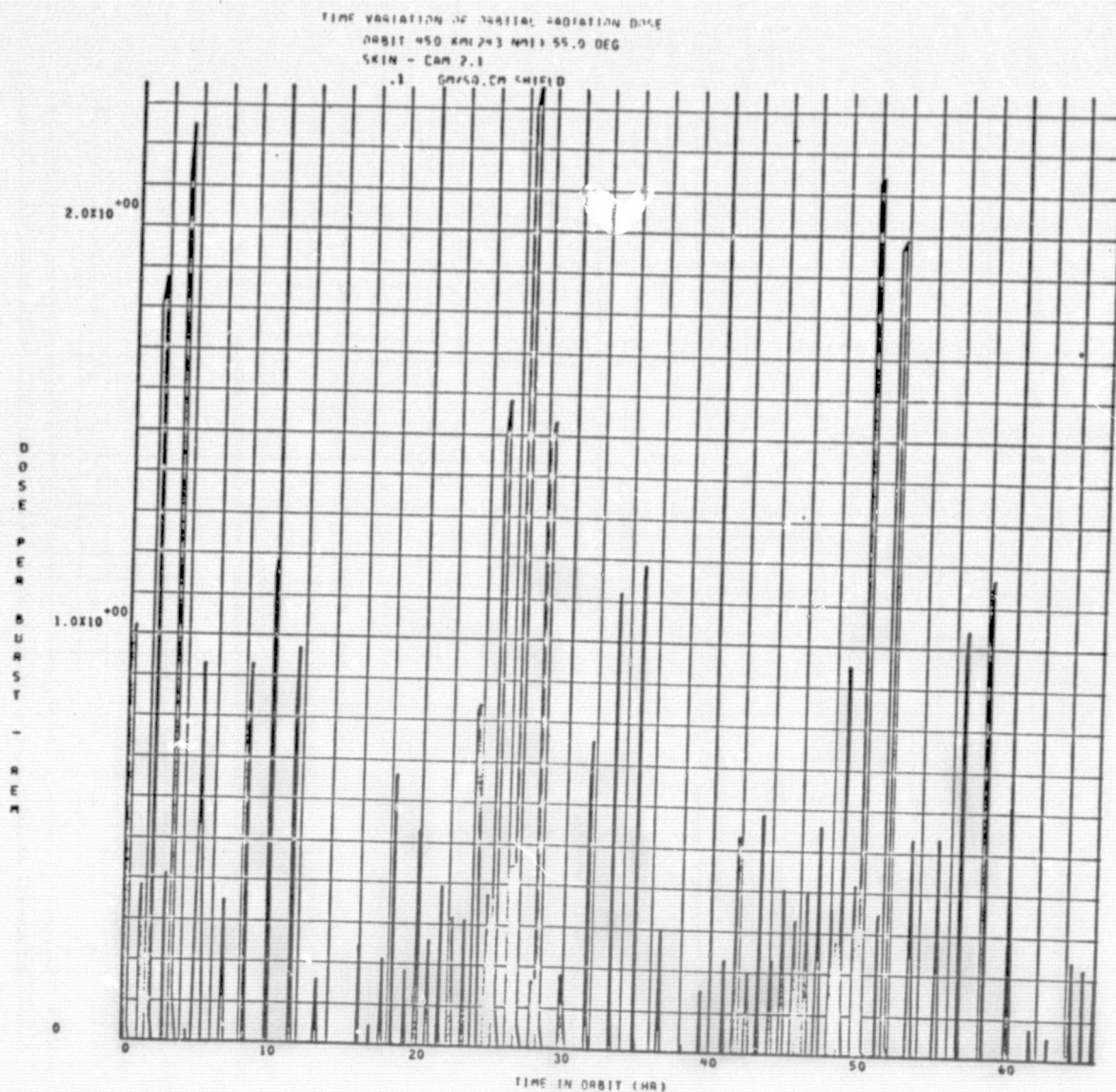


Figure 12-17. Time Variation of Orbital Radiation Dose Orbit 450 km (243 nmi) 55.0 Deg Skin - Cam 2.1

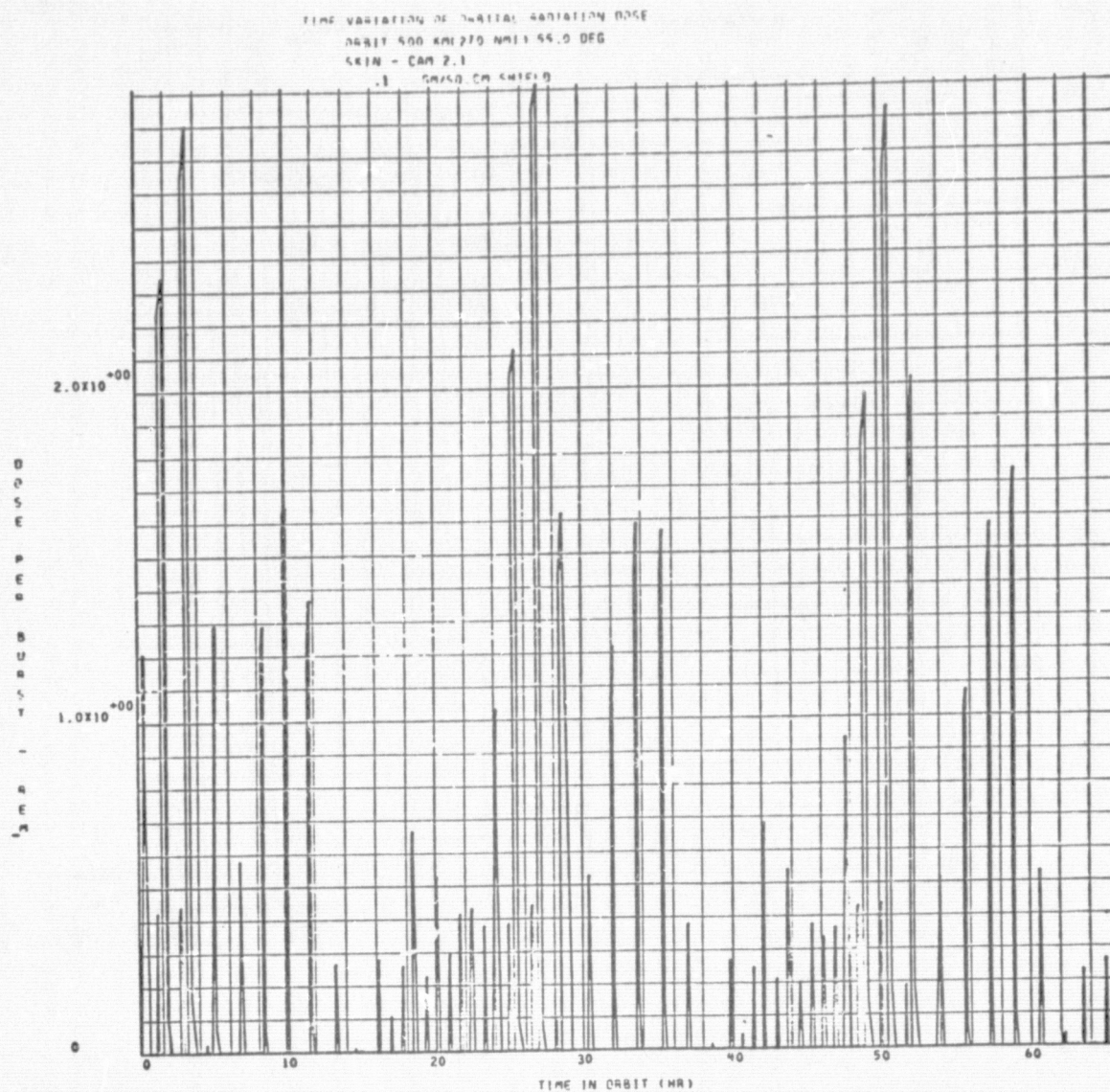


Figure 12-18. Time Variation of Orbital Radiation Dose Orbit 500 km (270 nmi) 55.0 Deg Skin - Cam 2.1

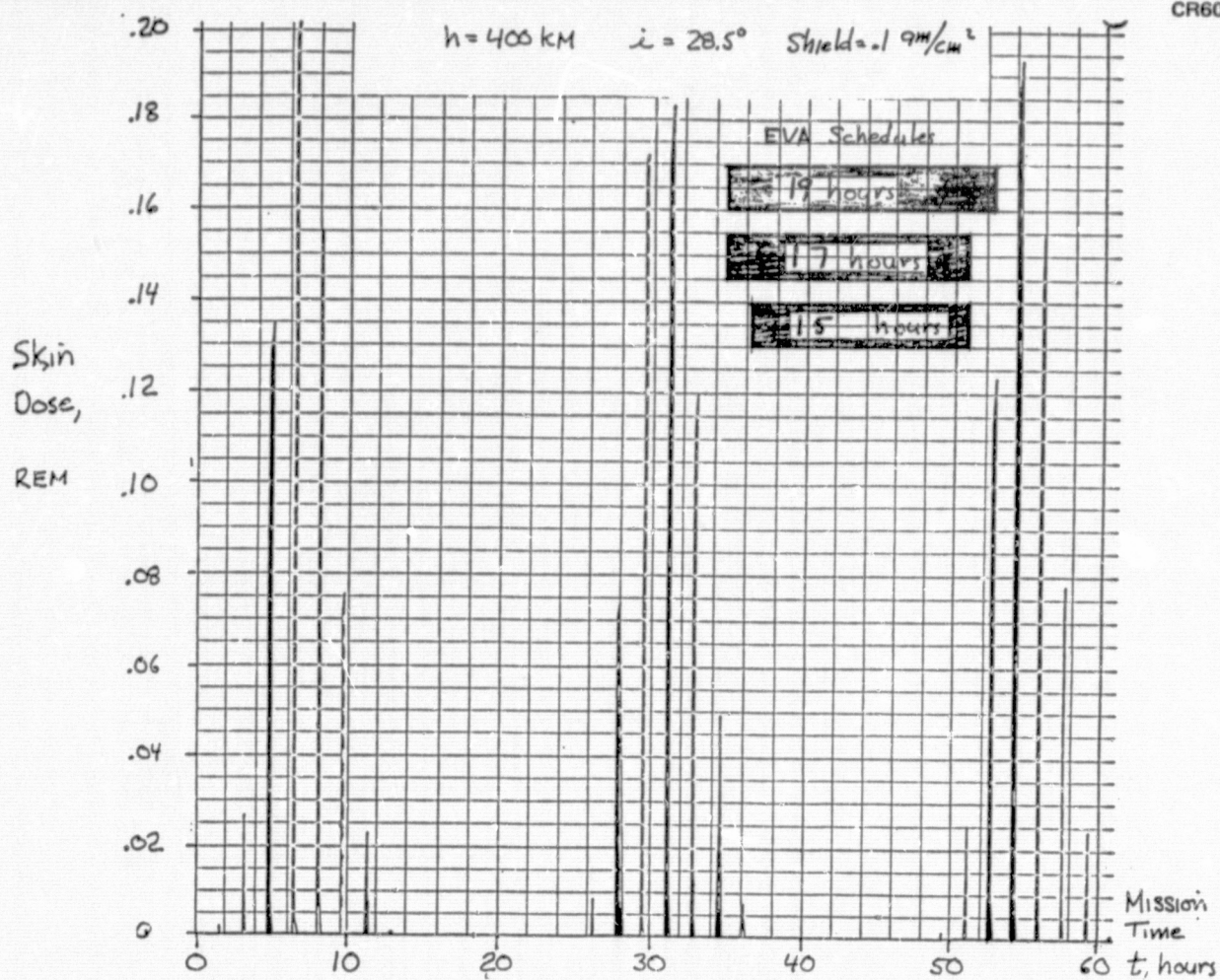


Figure 12-19. EVA Radiation Dose — Skin

The integration of the spike dose absorbed resulted in the EVA dose plots of Figure 12-20. This shows the daily EVA skin dose received as a function of orbit inclination, altitude, EVA shift duration, and EVA suit thickness. Also shown are the allowable dose percentages. Recall from Figure 12-3 that the Shuttle suit thickness is 0.1 gm/cm^2 and from Figure 12-20, large amounts of EVA could be accomplished on 28° missions while using only a small fraction ($\leq 10\%$) of the total allowable dose. The EVA capabilities shown for 28° , 15-17 hours, are in excess of the physical capabilities of the crew (about 6-8 hours per day). At higher inclination, the dose received is much larger even for shorter EVA times and thicker suits. To keep the skin dose down to $\sim 50\%$ of the allowable, a shift duration of much less than six hours per day using the Shuttle suit would be allowed for daily EVA activities. Shorter shift durations were examined for the 55° case as shown in Figure 12-21 where the EVA hours per man capability as a function of shift duration and crew sizes and shifts is shown.

This 55° data is plotted for the conditions shown keeping the EVA dose to 100 REM or less. For short EVA shift duration (2-4 hours) the 100 REM is not limiting and the available hours are the multiple of the shift duration times 180 days. As the shift duration increases (> 5 hours) the 100 REM limit is marked prior to 180 days. Very few shifts of 8 hour duration produce the 100 REM limit. (Recall the increasing number of radiation spikes as shift duration increases in Figure 12-19.) The EVA time for 180 days is maximized at a shift duration of about six hours. Adding a second crew (working a non-overlapping shift), increases the maximum time by only 23% because the second crew must find shift times in a higher radiation density portion of the day.

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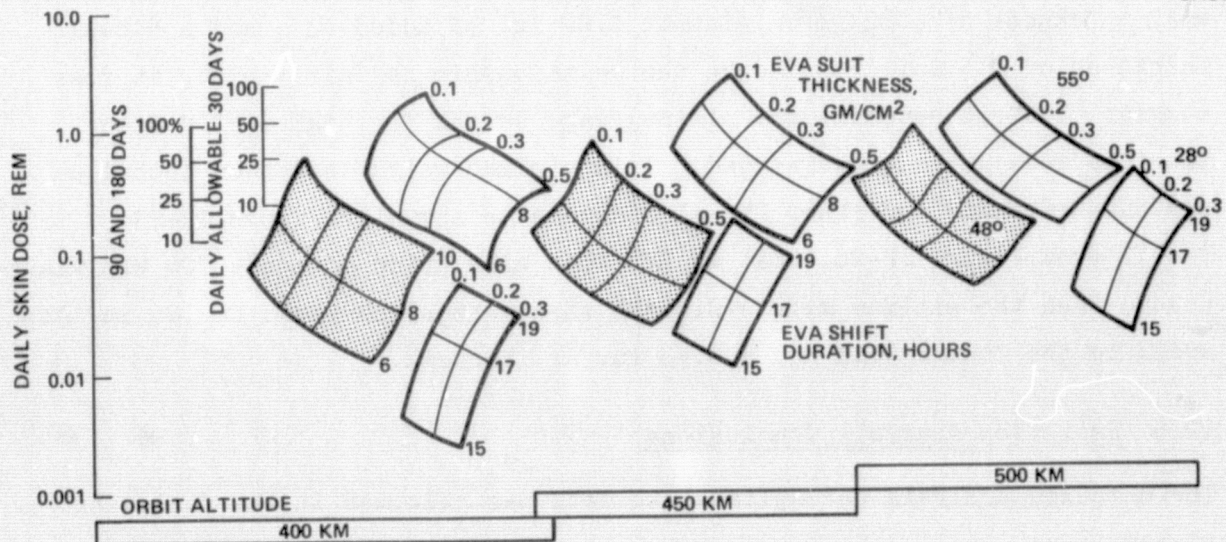


Figure 12-20. EVA Daily Skin Dose

Splitting the shifts (i.e., each crew works two short shifts per day) is illustrated by the 4 shifts, 2 crew curves. It provides a good means for getting the hours up for short shifts, however, with the preparation and suit donning and doffing time added, the single shift of 6 hours would be more efficient. The EVA requirement ranges in the right portion of Figure 12-21 were derived from the individual objective elements assuming two men per EVA crew. A 6 hour shift provides most of the hours needed to meet these requirements. The deficit could be made up by extending the time over which the EVA tasks were performed (to lower the daily dose) or providing thicker EVA suits.

12.4 Shield Calculations

Analysis of the module wall, biowell, and EVA suits shield requirements was accomplished by considering the three dose sources together.

At 28.5°, the standard module construction would provide adequate shielding from trapped radiation (~50% allowable dose - Figure 12-5), the magnetic field cuts off any SCR dose - Figure 12-6, and the Shuttle EVA suit would keep EVA dose to less than 10% of the allowable - Figure 12-20.

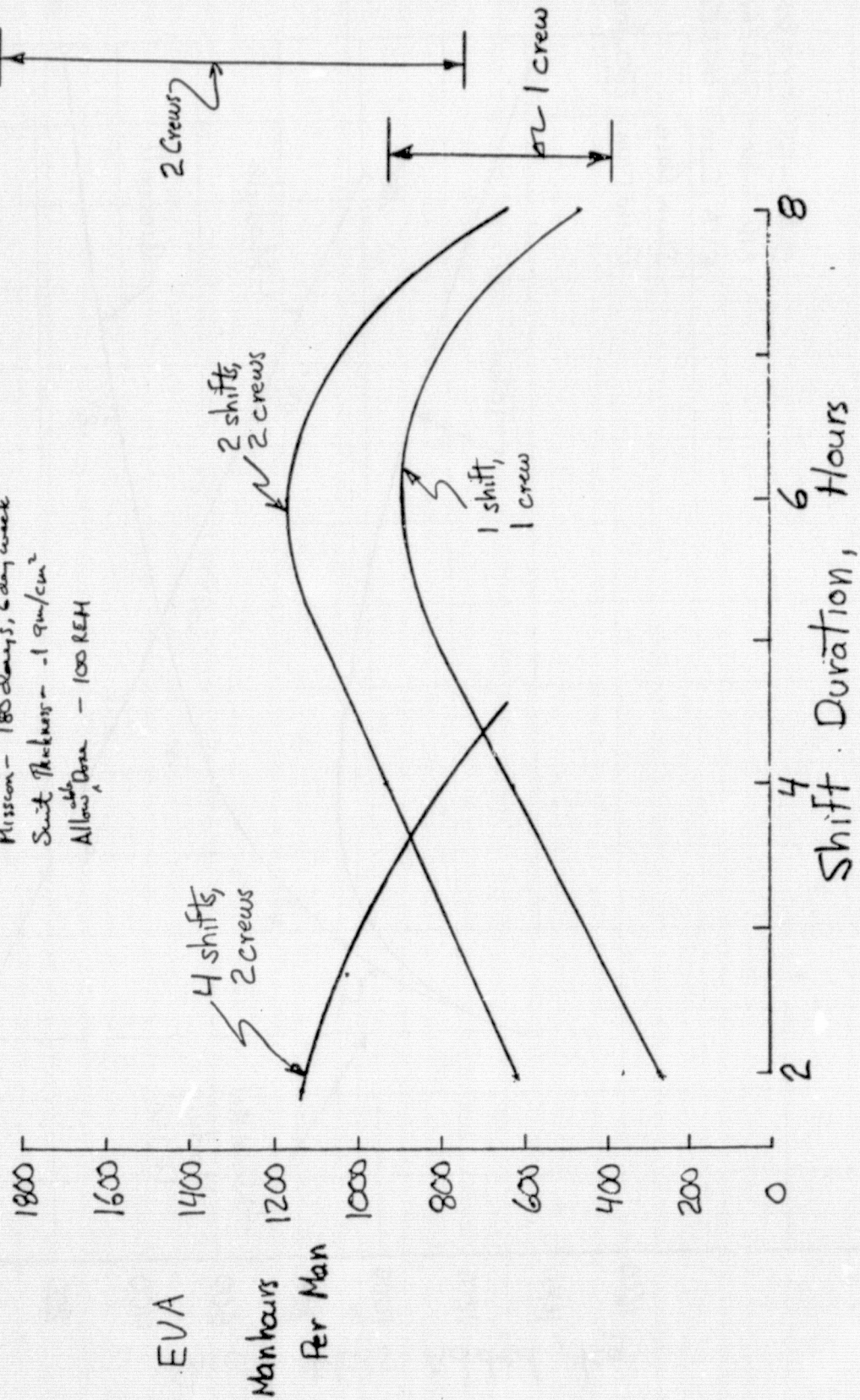
At 55°, increased module shielding, a biowell, and potential EVA restrictions are needed. A proper balance between these shield designs was determined. Figure 12-22 shows the module wall shield weight and biowell shield weight needed as a function of module wall thickness for a 55° orbit at 450 km. This relationship was drawn for a 180 day mission allocating 40 REM to EVA. As seen, the minimum shield weight solution for this case occurs at a module wall thickness of 0.82 gm/cm² (about 0.02 in. of added Al), and a Biowell thickness of 0.5 in. Al. Though the relationship is fairly flat, it does suggest that the penchant for minimum gage design for high inclination missions does not seem warranted. This minimum weight solution technique was then extended for other EVA allocations as shown in Figure 12-23. Recall from Figure 12-20, that an EVA dose allocation of about 100 REM was needed then the minimum module/Biowell shield required would be .06 in. Al added to the module wall and a 0.43 in. Al. Biowell.

12.5 Radiation Analysis Conclusions

The detailed analysis accomplished to date has resulted in the following conclusions:

EVA Reqmt Range

Orbit - $55^\circ \times 450 \text{ km}$
 Mission - 180 days, 6 day week
 Suit Thickness - 1 gm/cm^2
 Allowance - 100 REM

Figure 12-21. EVA Capability - 55°

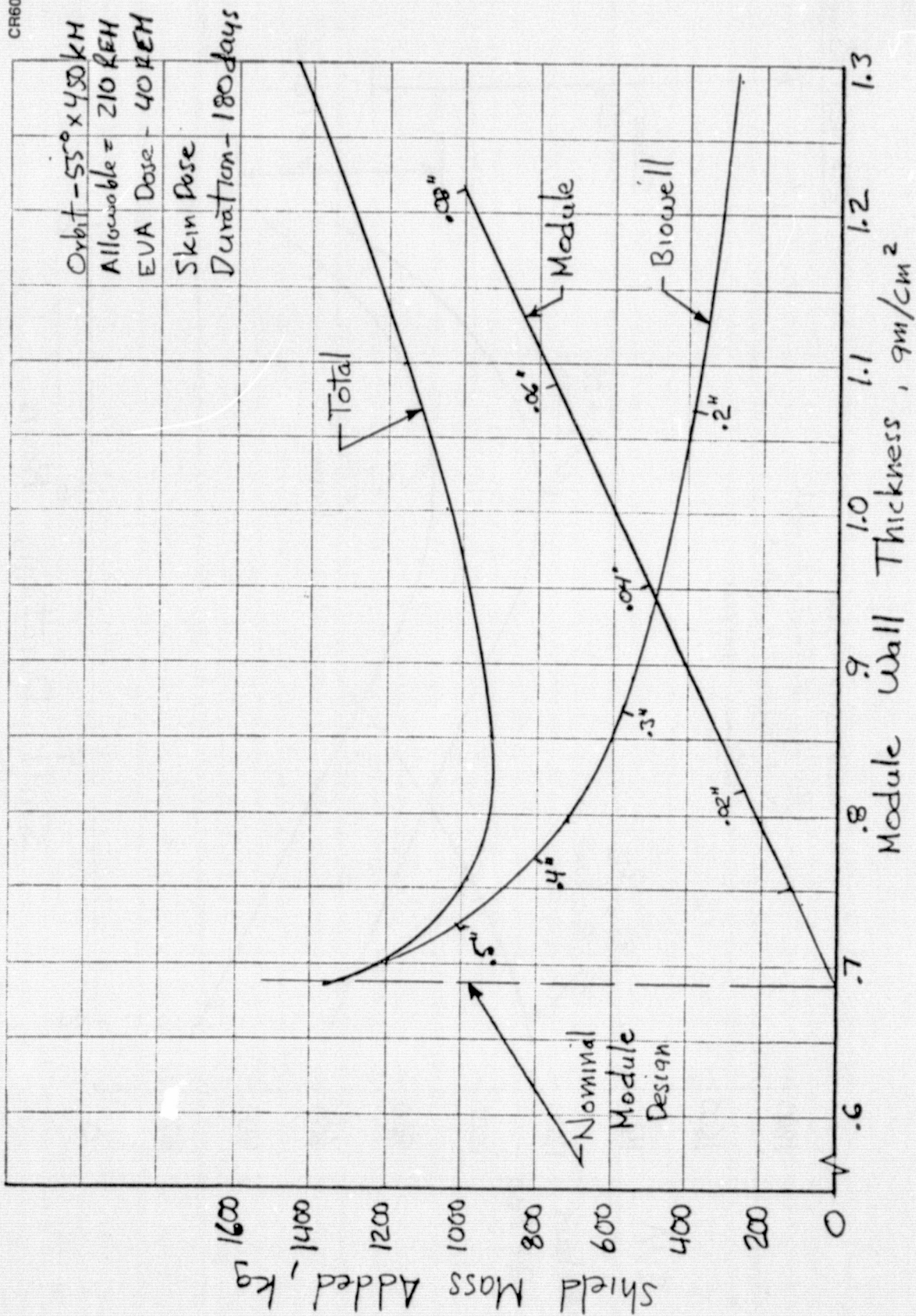


Figure 12-22. Module/Biowell Shield Optimization

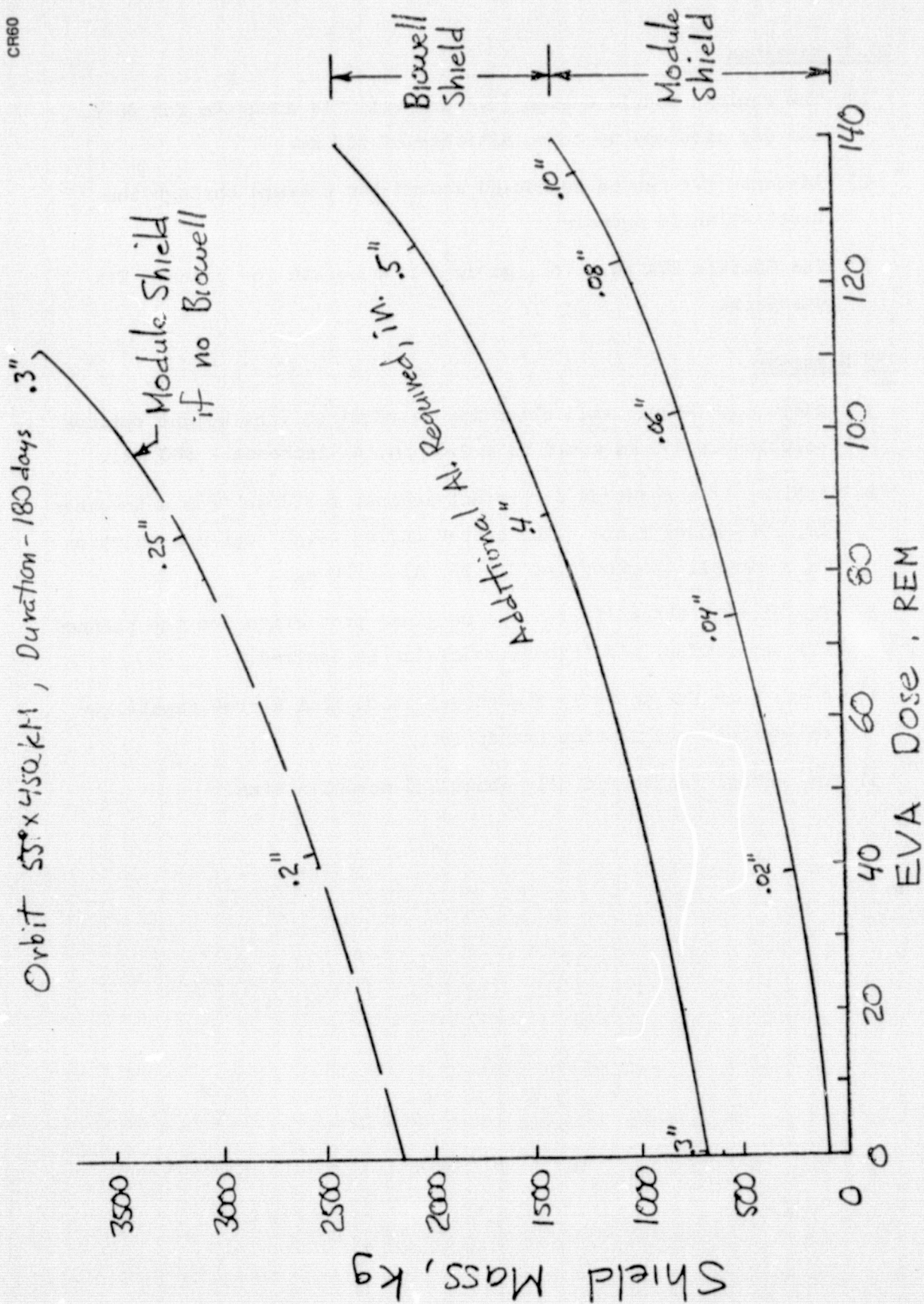


Figure 12-23. Minimized Module/Biowell Shield Required

28.5° Missions

- 1) The nominal module design (0.1 in. wall) is adequate for 90 to 180 day missions up to an altitude of 560 km.
- 2) Adequate EVA can be scheduled around the passage through the South Atlantic anomaly.
- 3) The Shuttle EVA suit (0.1 gm/cm^2) is adequate for planned EVA durations.

55° Missions

- 1) Additional module wall shielding is required (the weight optimum solution at 450 km would be a 0.06 in. Al increase - 800 kg).
- 2) A Biowell is required to protect against the dose from a November 12, 1960 model flare. The corresponding weight optimum solution is a Biowell thickness of 0.4 in. Al - 700 kg.
- 3) The Shuttle EVA suit provides marginal protection for the planned EVA activities. Additional shielding is desired.
- 4) A six hour EVA shift is a good selection from a crew capability and maximum utilization standpoint.
- 5) The second shift add only about 23% more EVA time.

Section 13

SPS ORBITAL TEST REQUIREMENTS

13.1 TA-1L TEST PLAN

The test plan proposed for Test Article 1, LEO (TA-1L), with the antenna consisting of two crossed line sources one 123m and the other 125.6m long would explore the following issues:

- A. Validate the theoretical beam steering accuracy of the crossed array.
- B. Validate the theoretical sidelobe distribution predictions by means of LEO measurements (line source measurements only).
- C. Validate the dynamic retro-directive steering capability (pilot beam steering) of the crossed array.

The total number of separate test setups proposed for the TA-1L is 29.

Sidelobe measurements will be performed on the horizontal (prototypical) line source array. For all the proposed tests, the line source will be rotated around the yaw axis $\pm 5^\circ$ away from broadside. This angular rotation will allow investigation of the antenna pattern out to the first subarray pattern sidelobe peak.

As Figure 13-1 shows, the test tree is broken down into two subsets. The second subset (tests 8 through 11), with the line source illumination tapered as developed in Part II of the study, would measure sidelobe variations as a function of amplatron input voltage variations; in addition, one of the voltage variations would also be combined with varying the source or the pilot frequency.

In the first subset (tests 1 through 7), those tests are included which will show greater sensitivity with the line source being uniformly illuminated. These include measuring the line source's sensitivity to (mainly) power tube induced temperature variations, mechanical distortions (see Figure 13-2),

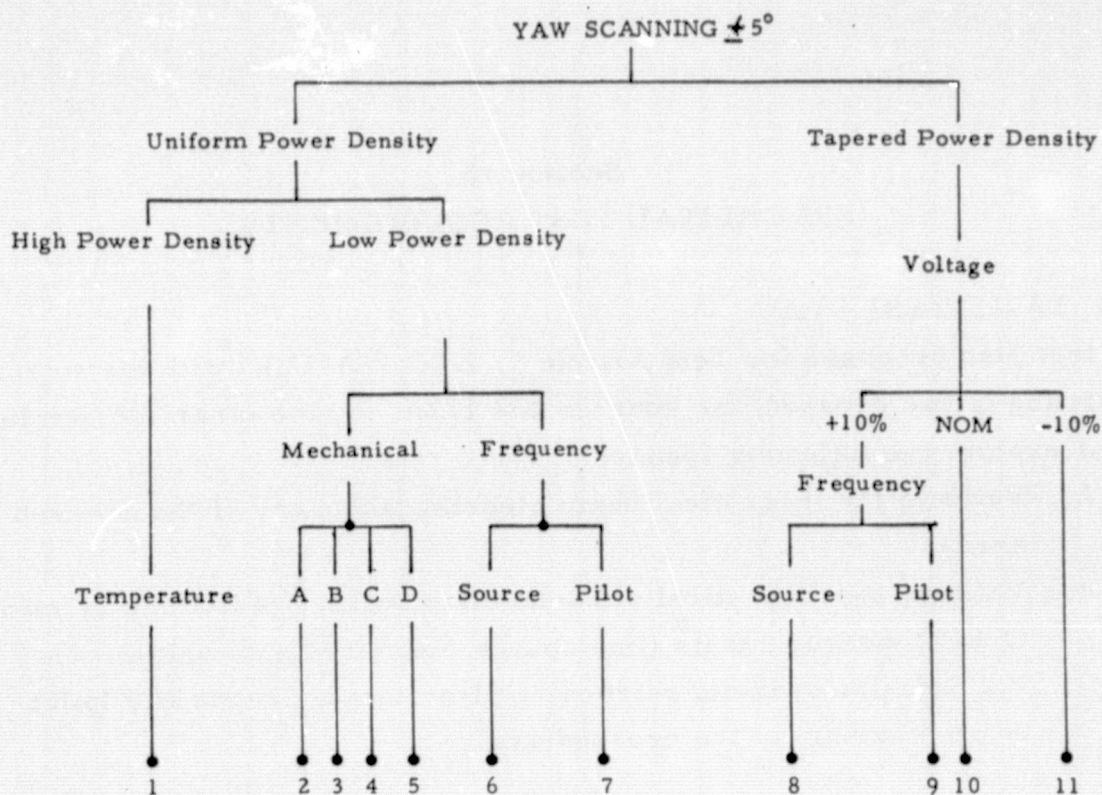


Figure 13-1. Sidelobe Measurements

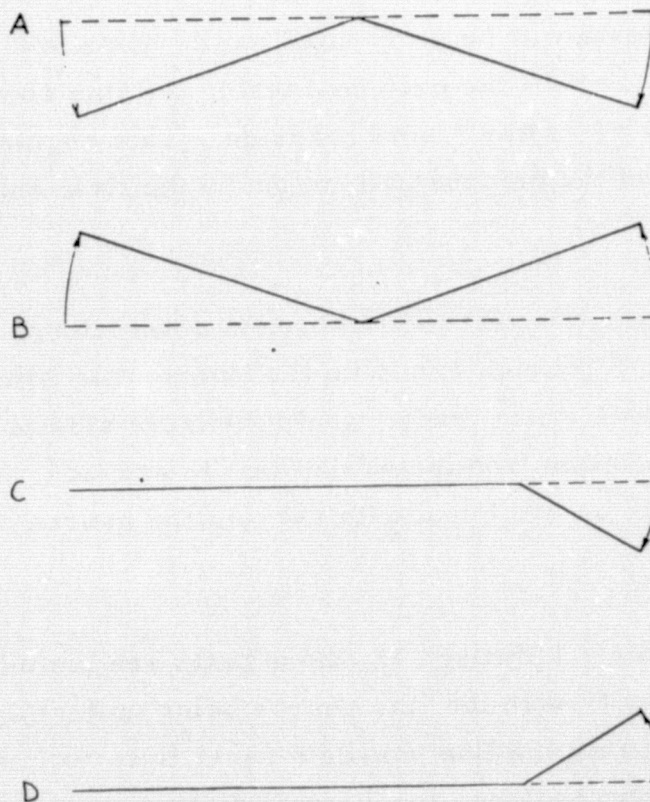


Figure 13-2. Mechanical Distortion of Linear Array

and sidelobe changes introduced by frequency changes in either the pilot or the source frequency. The uniform power density subset resulting from this test plan represents a potential requirement for a more elaborate TA-1 antenna than the tapered version of Part II. The changes, which require further study, would likely be accomplished by on-orbit reconfiguration of the microwave electronics (waveguides, amplifiers, and phase control). The high power density test (test 1) should perhaps be deferred to TA-2.

The beam steering accuracy tests are broken down into three subsets; (see Figure 13-3): (1) static patterns; (2) phase control accuracy, and (3) frequency scanning effects.

Steering of the beam under any of these tests would be performed over a $\pm 1^\circ$ field of view (i. e., over a field of view of ± 5 beamwidths).

For the static patterns, beam steering accuracy tests will be performed (in the plane of the line source) for the horizontal array and the vertical array; a third test will consist of investigating the beam steering accuracy of the combined array in the diagonal plane.

The effects of phase control accuracy on beam steering accuracy will be measured. The effects of both electronic and mechanical phase control distortions will be measured.

The third set of tests under beam steering accuracy will investigate the effects of source beam/pilot beam frequency changes.

The last set of tests involves the measurement of dynamic retrodirective steering (see Figure 13-4). In these tests, measurements will be taken in four planes spaced 45° apart. The array will be rotated in each one of these planes by $\pm 5^\circ$, and the phase commands generated for each subarray will be recorded. The shape of the beam will be recorded at the center frequency. Then the frequency of the pilot or the source will be changed, and the resulting phase commands and beam shapes will again be recorded.

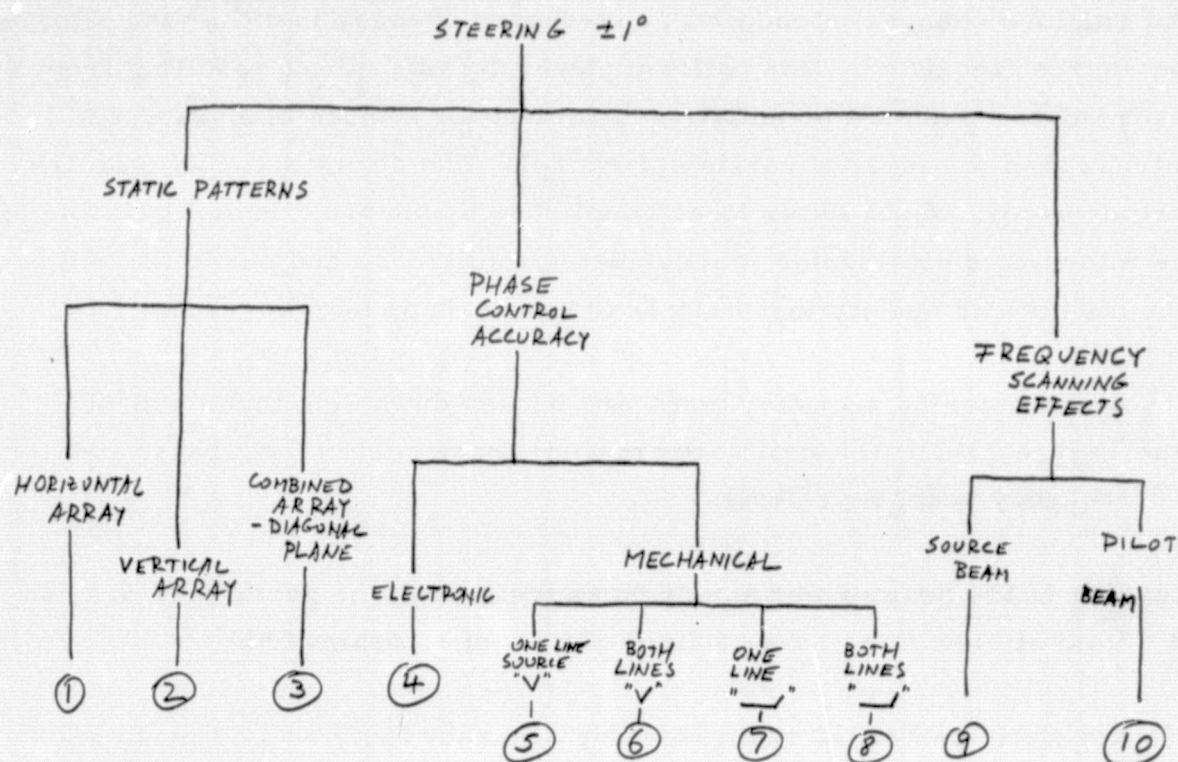


Figure 13-3. Beam Steering Accuracy (Tapered Crossed Arrays)

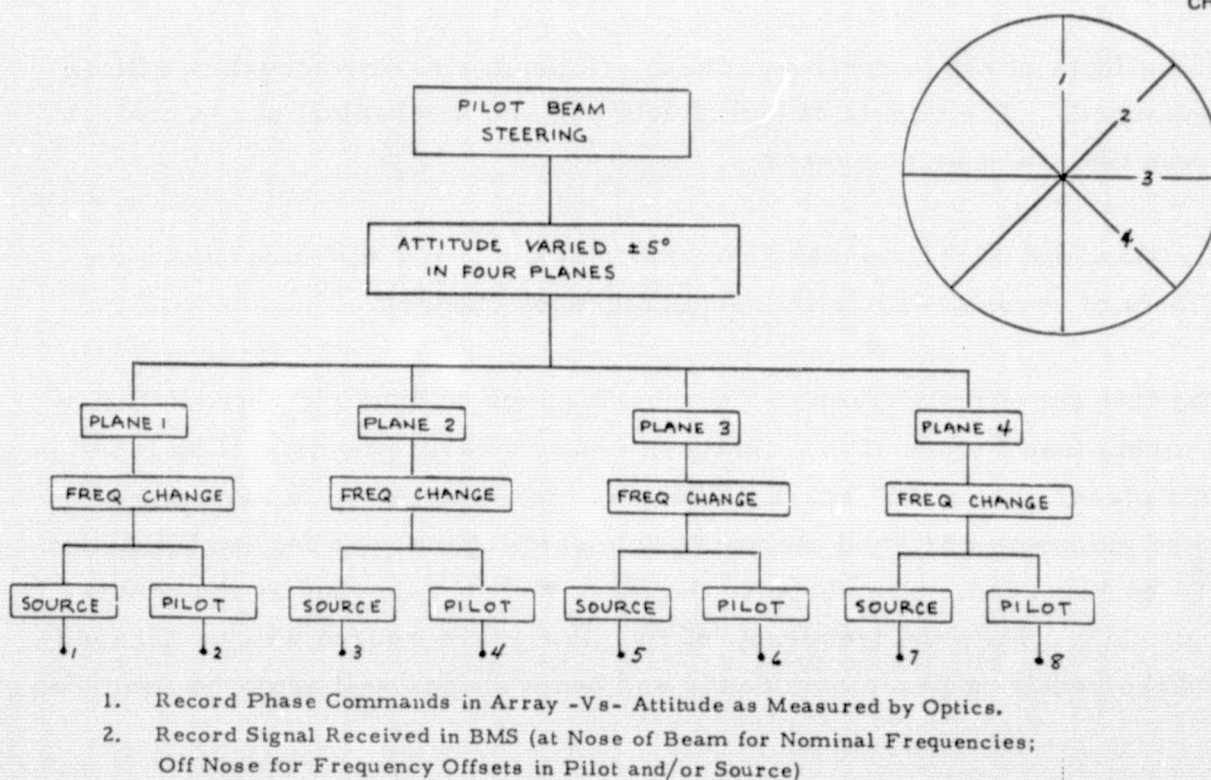


Figure 13-4. Dynamic Retrodirective Steering

13.2 TA-2 ANTENNA MEASUREMENT

The SPS test article array antennas are focused on the mapping satellite by the pilot signal. Consequently, a question arises as to the necessity of separating the mapping satellite from the test article by $2D^2/\lambda$ as normally required on an antenna pattern range, where the array antennas are focused at infinity. To resolve this question, the proposed TA-2 antenna configurations were modeled in a ray-tracing program to ascertain the radiation characteristics at the satellite under various ranges, weighting and focusing.

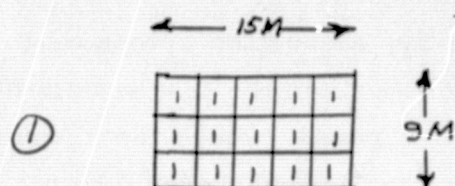
The antenna configurations analyzed are shown in Figure 13-5. The subarray size employed in focusing was 1.5 by 1.5 meters. The sector size used for weighting was 3 by 3 meters as shown in Figure 13-5.

Figures 13-6 through 13-9 show the impact of focusing and weighting on the received main lobe at the $2D^2/\lambda$ range of 3368 meters. These figures are first quadrant profile plots; thus, the main lobe covers twice the distances indicated on the figures. Even at this range, the unfocused configuration exhibits some first side lobe interference with the main lobe preventing a good null.

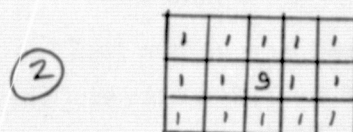
Figures 13-9 through 13-12 show the pattern contours obtained for a weighted and focused TA-2 configuration as a function of the mapping satellite range. As the distance decreases from $2D^2/\lambda$ the first side lobe merges with the main lobe, reducing the null depth, and at $0.5D^2/\lambda$ the first sidelobe is well within the main lobe area causing serious pattern deformity.

This phenomenon is more easily visualized on Figures 13-13 through 13-16 where the main lobe pattern amplitude along the X-axis is plotted in decibels.

System errors due to environmental conditions, stabilities and steering command can be measured and evaluated with more sensitivity by monitoring the amplitude of the grating lobes. The side lobe interference with the main lobe shown in the above figures also occur in the grating lobe areas. Furthermore, the smaller grating lobe amplitude will result in a more severe distortion. Thus, to obtain accurate error data, it is vital that the mapping satellite be separated by at least $2D^2/\lambda$.



TA-2 UNIFORM WEIGHT



TA-2 WEIGHTED

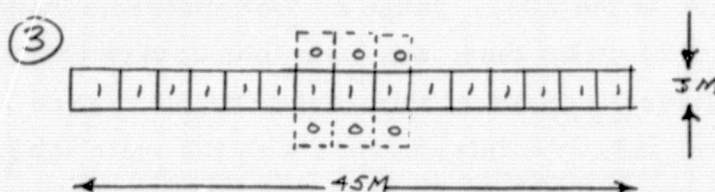
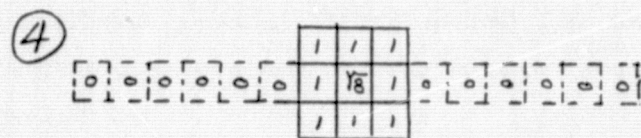
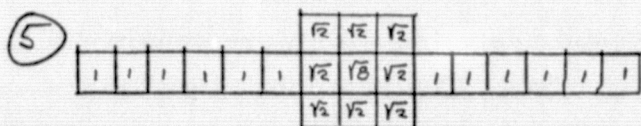
TA-2 ALTERNATE
(UNIFORM WEIGHT)TA-2 ALTERNATE
(WEIGHTED CENTER)TA-2 ALTERNATE
(FULLY WEIGHTED)

Figure 13-5. Weighting Configurations

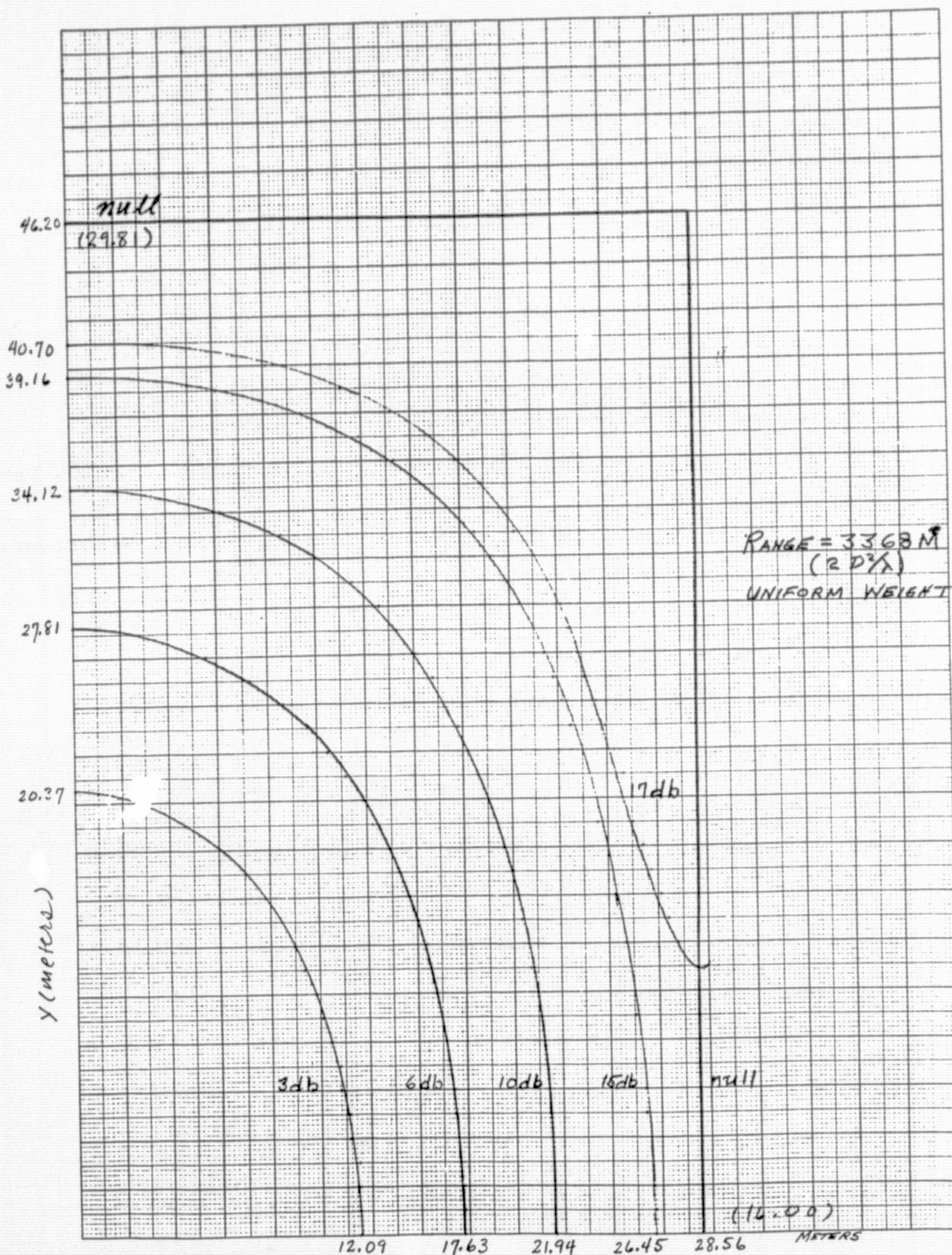


Figure 13-6. TA-2 Pattern, Focused at Infinity

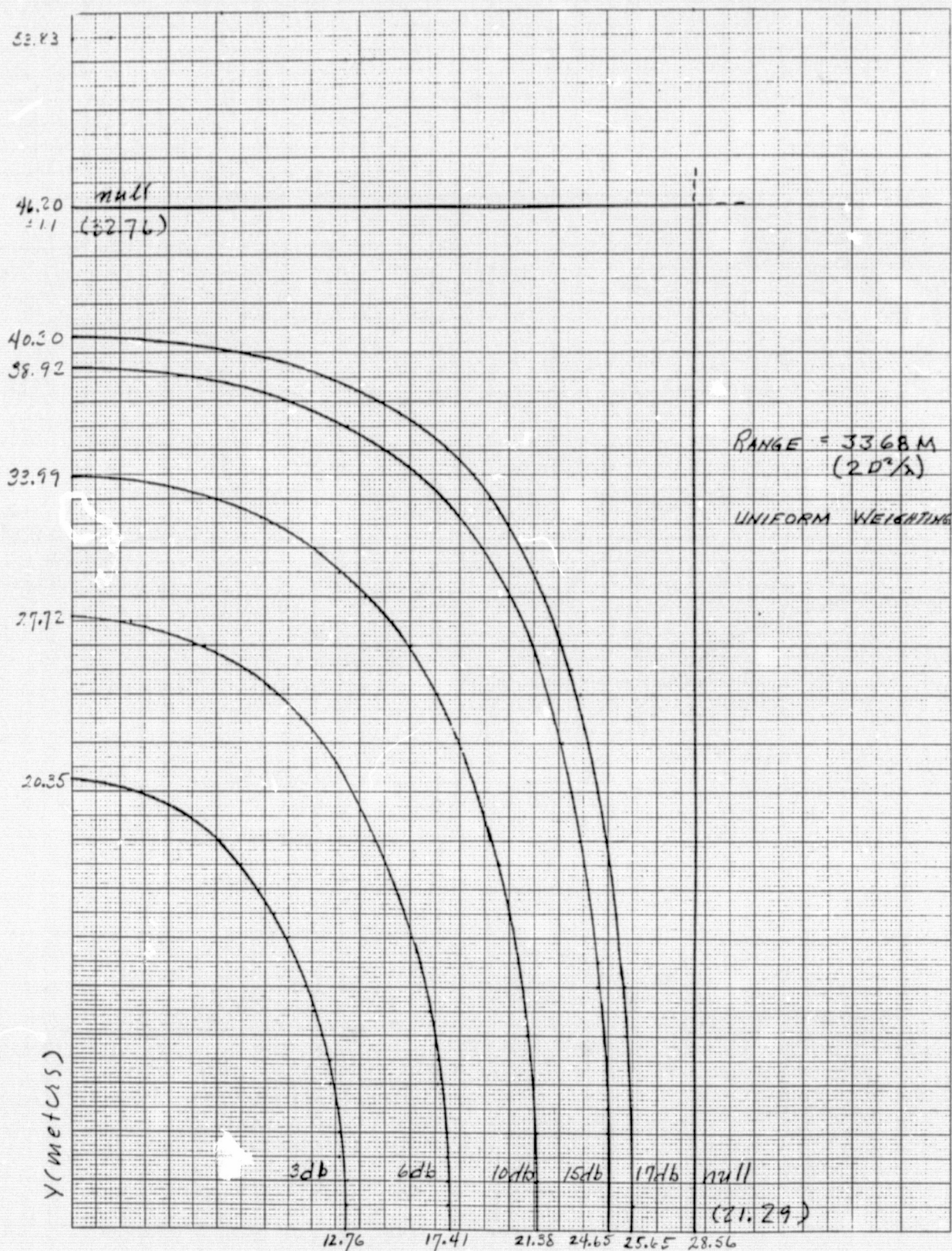


Figure 13-7. TA-2 Pattern - Focused at Satellite

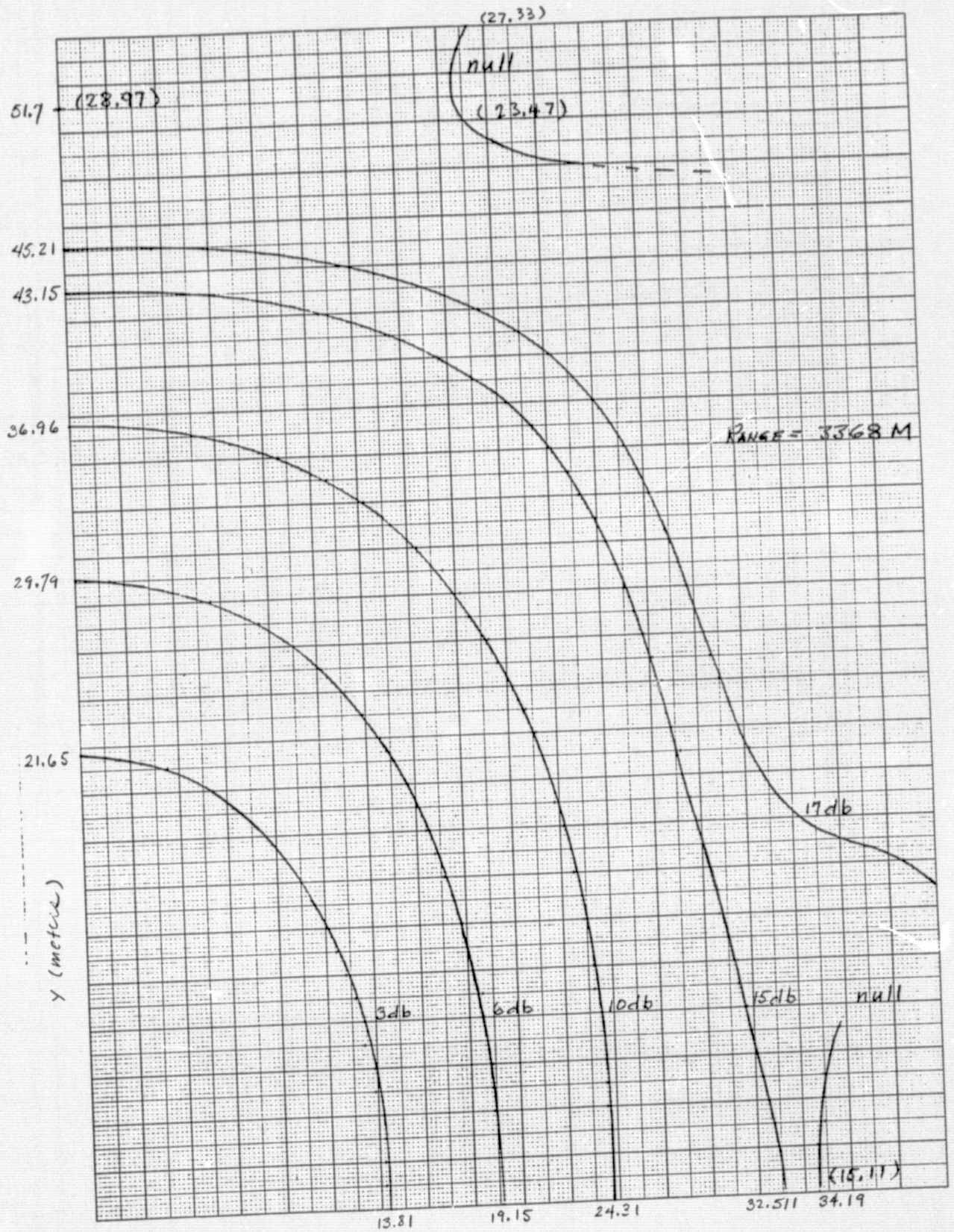


Figure 13-8. TA-2 Pattern - Weighted Without Focus

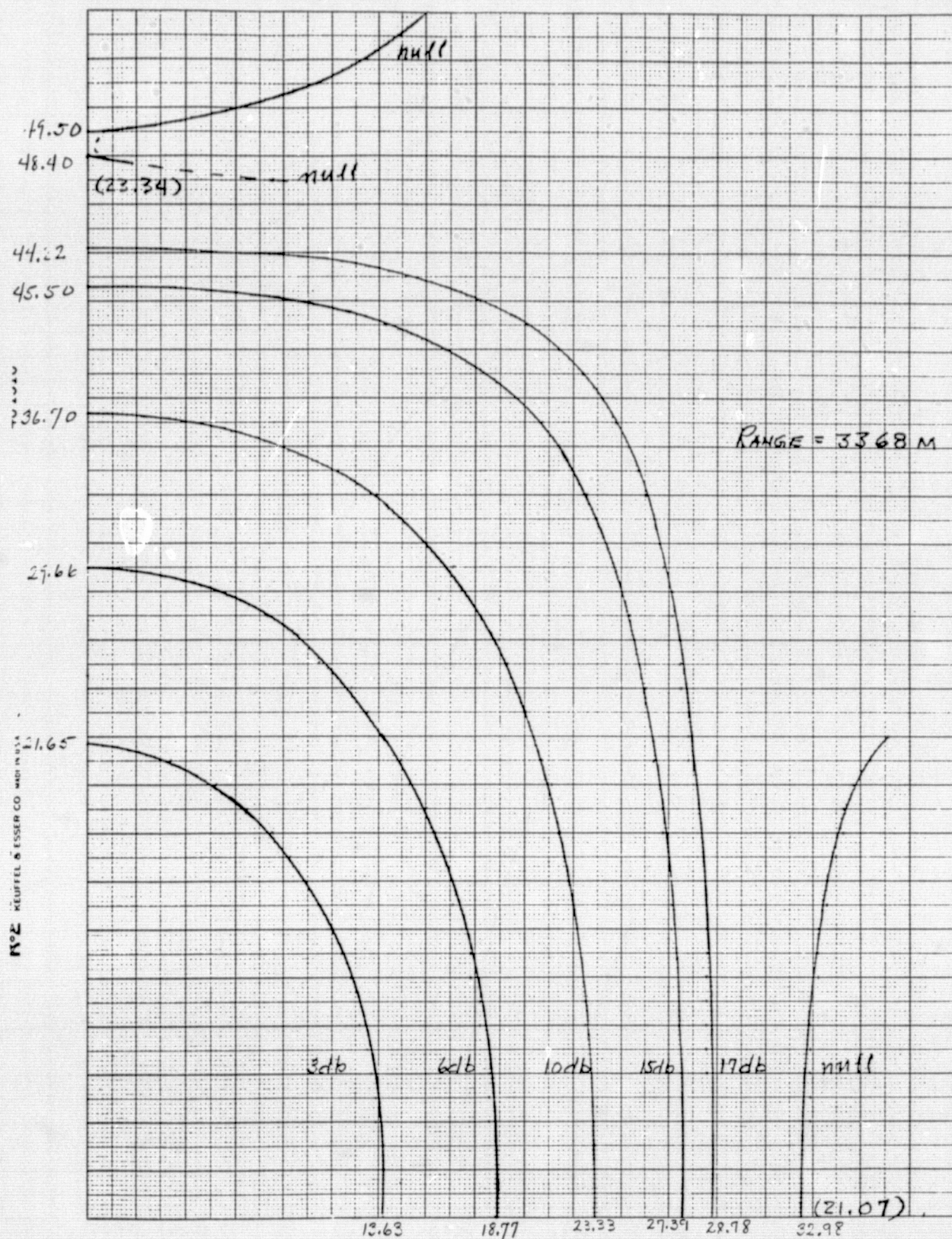
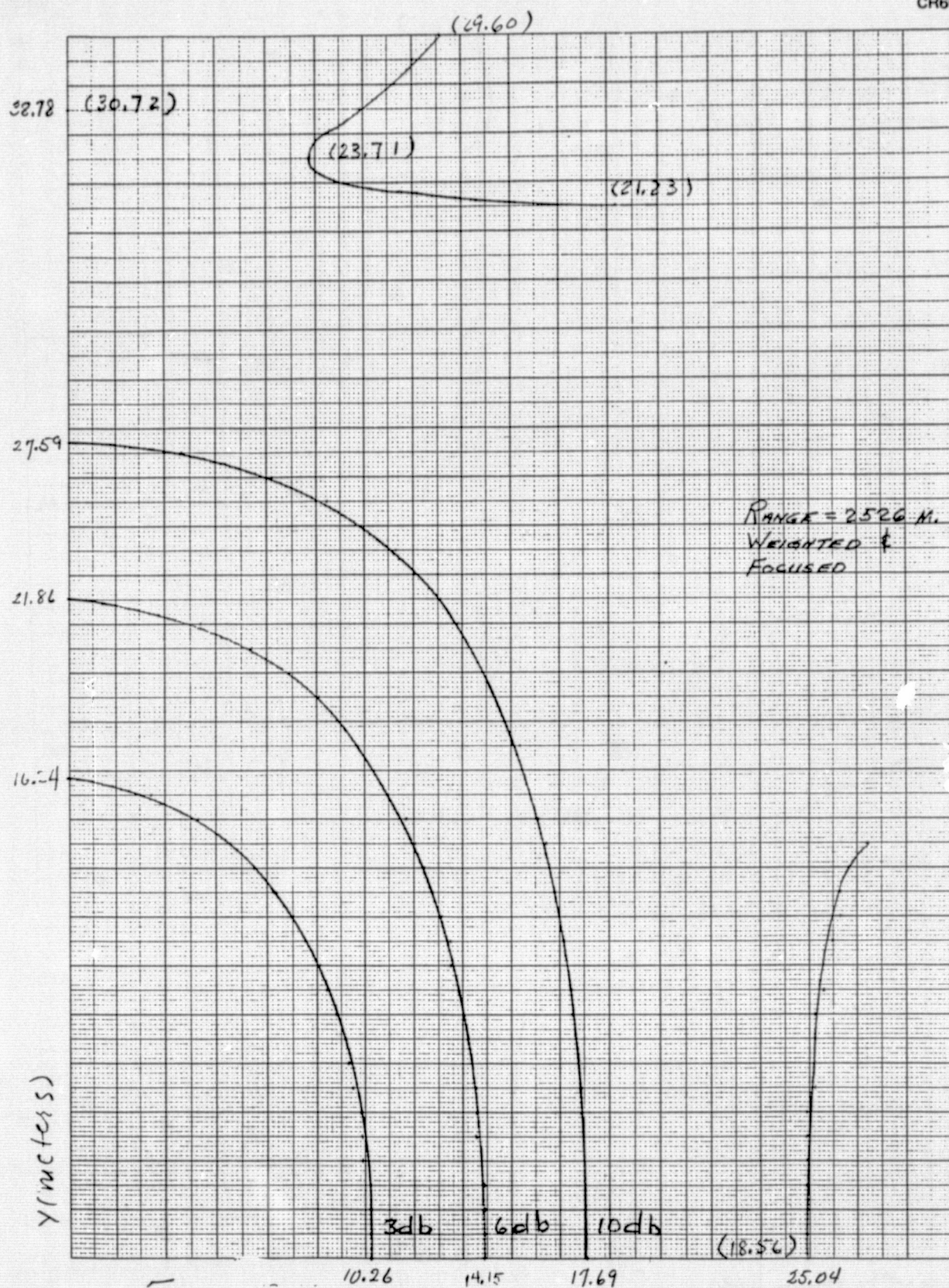
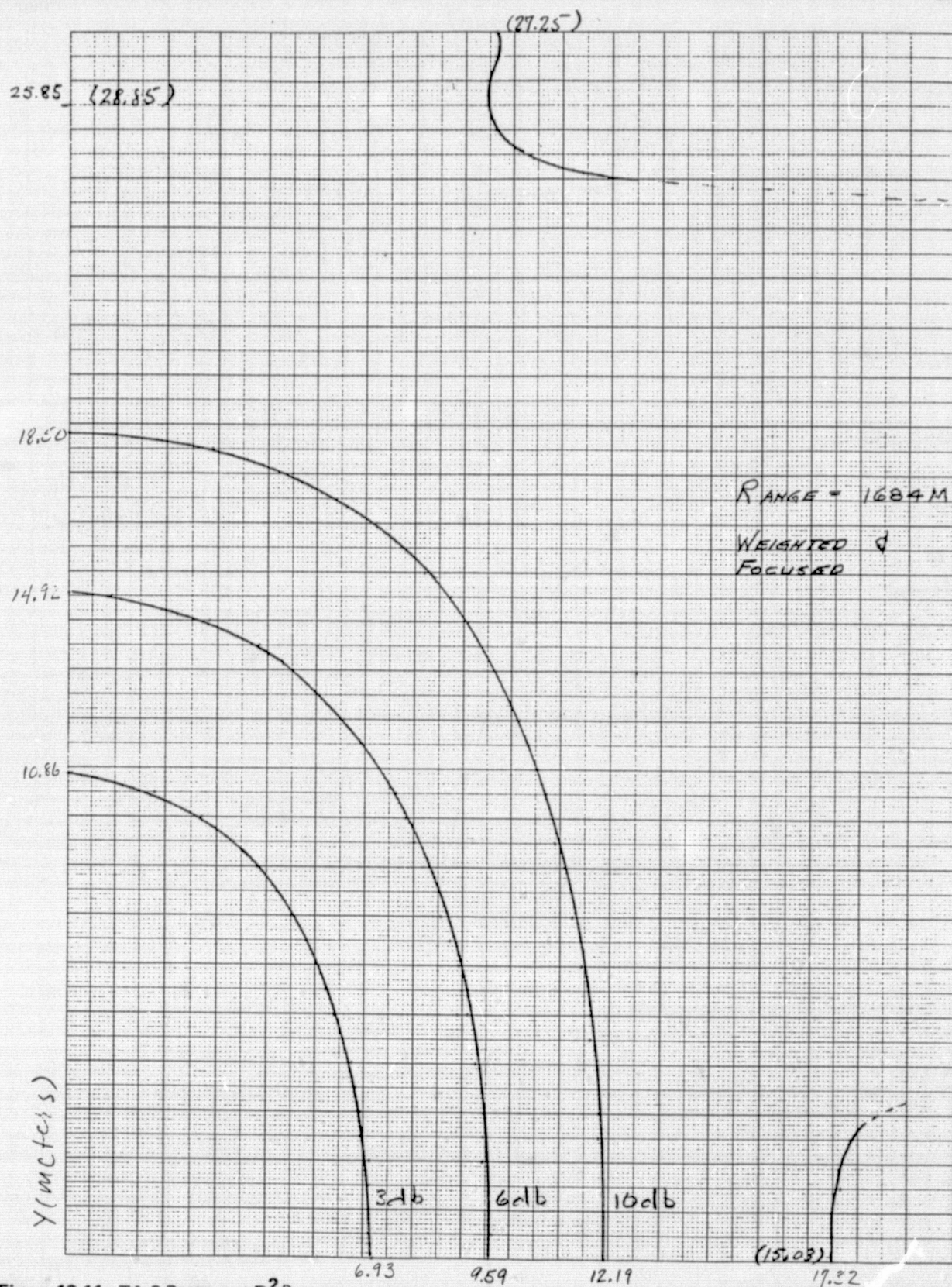
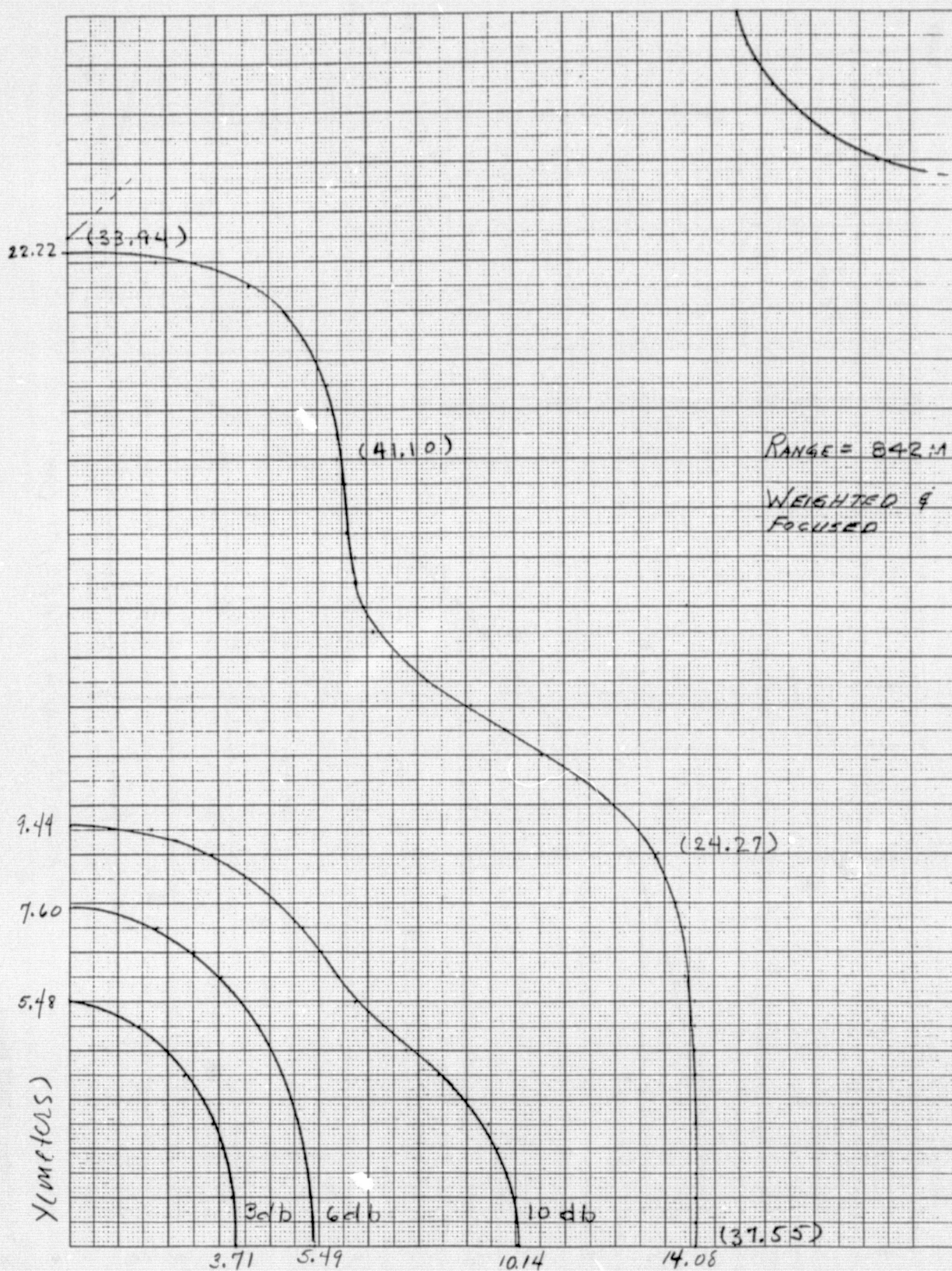


Figure 13-9. TA-2 Pattern - Weighted and Focused

Figure 13-10. TA-2 Pattern at $1.5 D^2/\lambda$

Figure 13-11. TA-2 Pattern at D^2/λ

Figure 13-12. TA-2 Pattern at $0.5 D^2/\lambda$

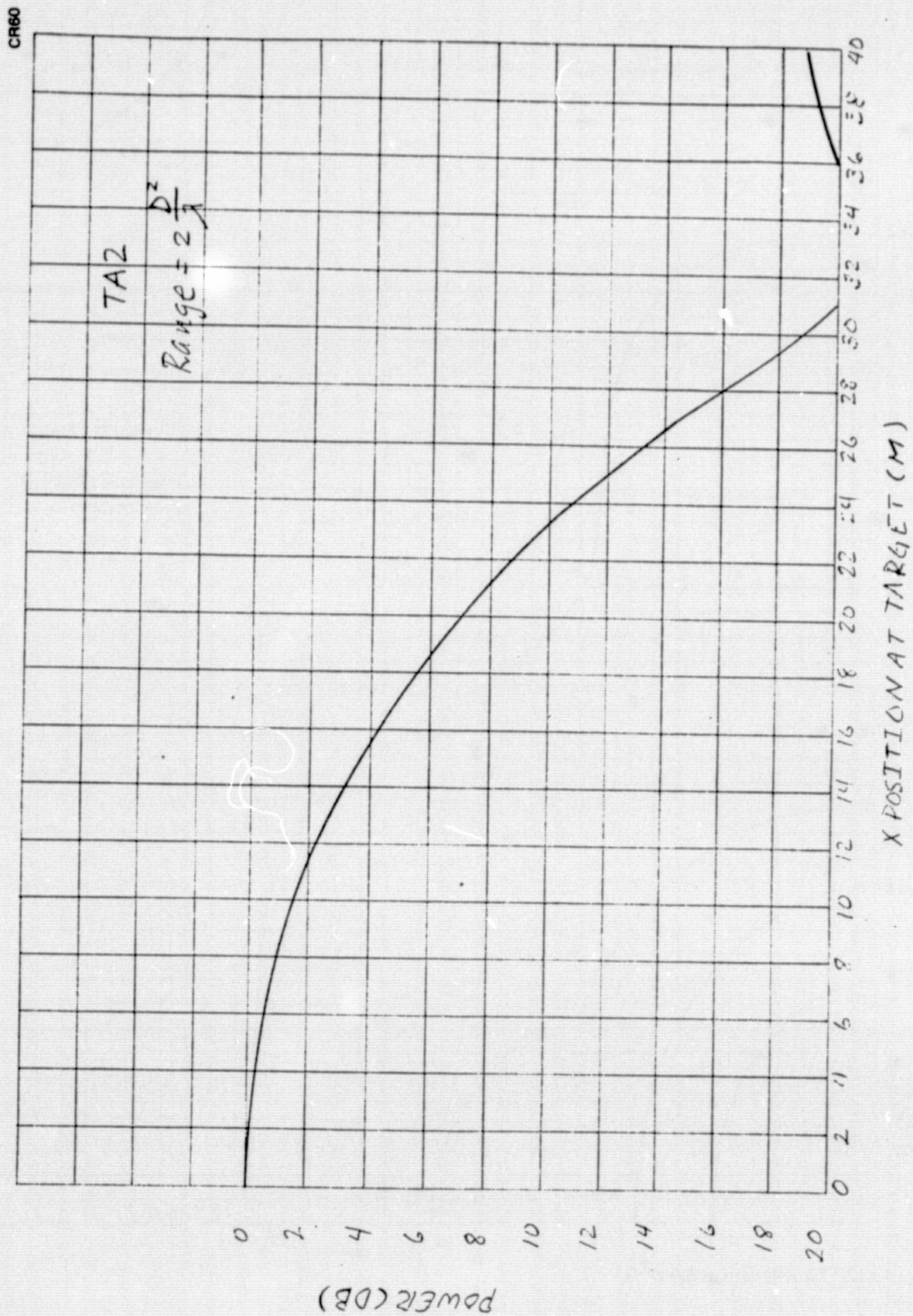


Figure 13-13. TA-2 Pattern at 3368 Meters

CF60

TA2
Range = $\frac{3D}{2\lambda}$

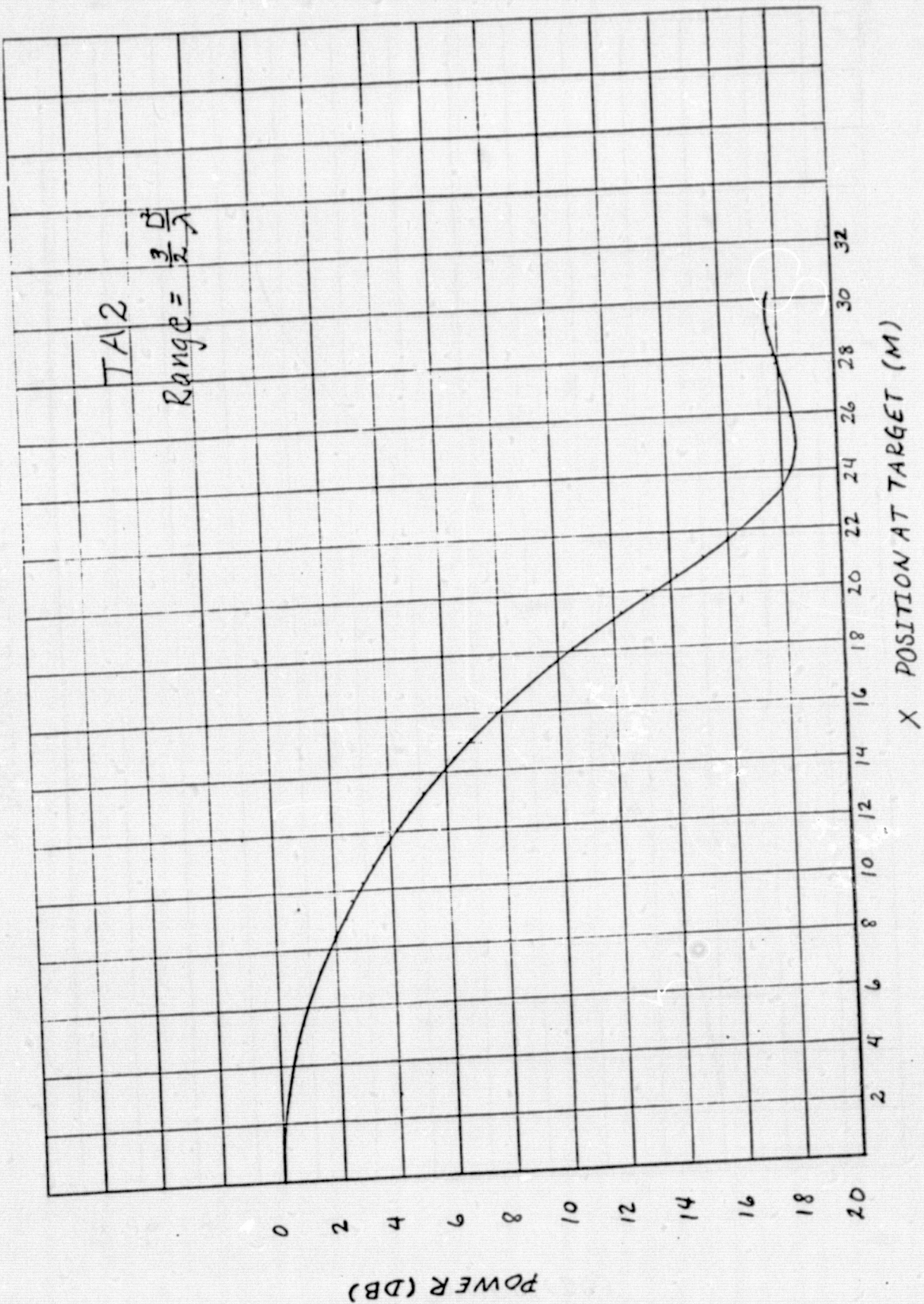


Figure 13-14. TA-2 Pattern at 2526 Meters

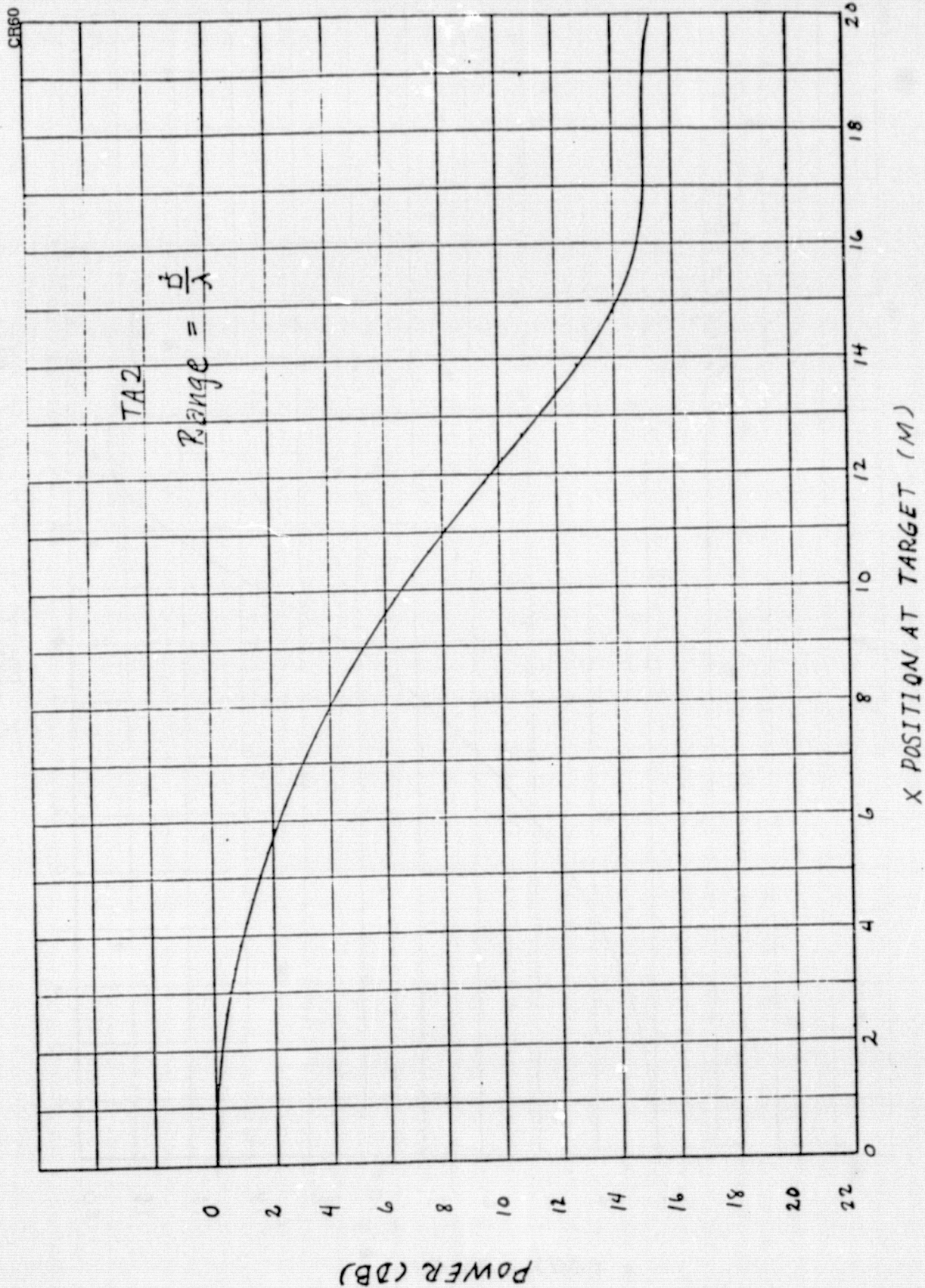


Figure 13-15. TA-2 Pattern at 1684 Meters

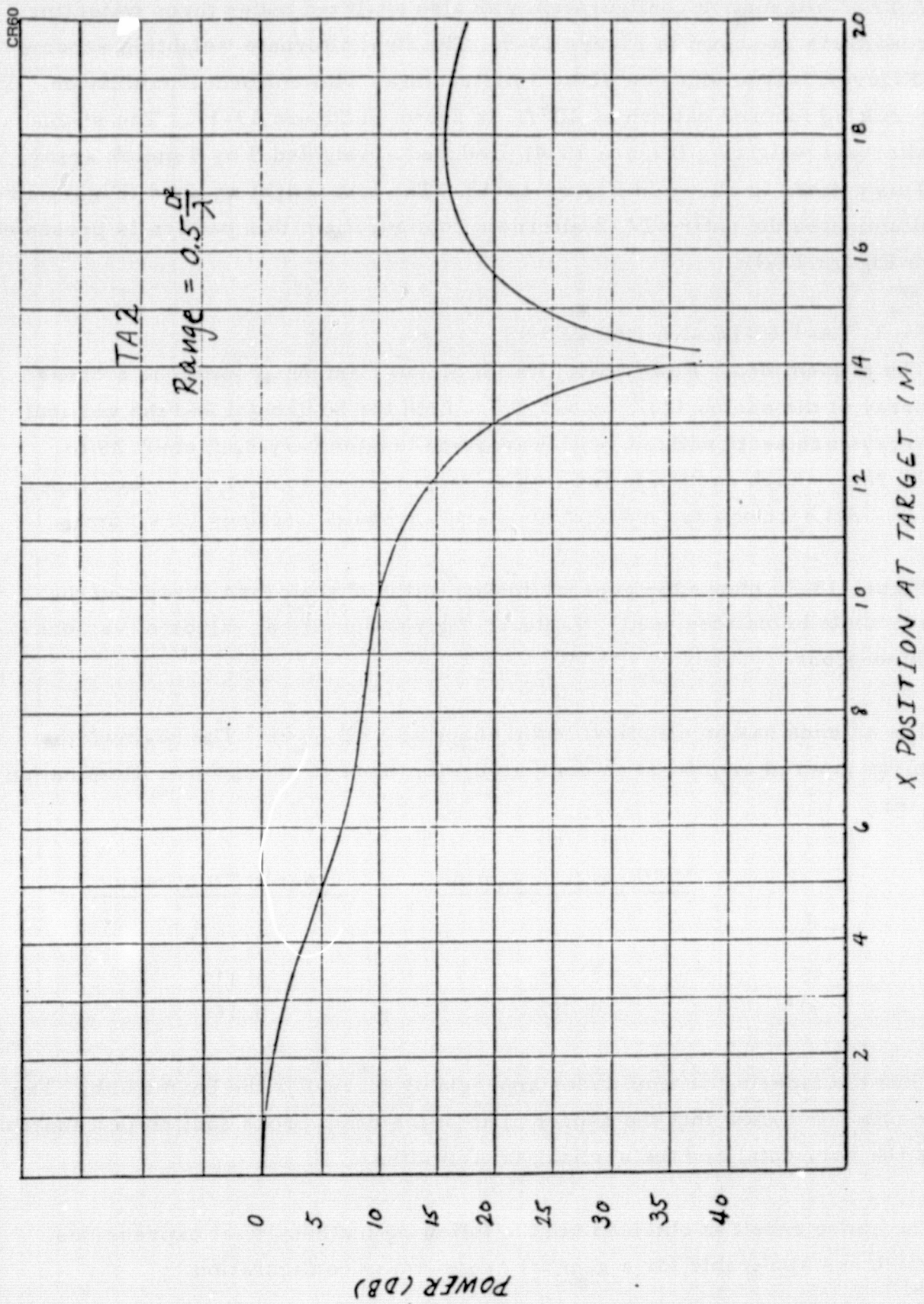


Figure 13-16. TA-2 Pattern at 842 Meters

A TA-2 alternative configuration was also analyzed under three weighting conditions as shown in Figure 13-5. The first alternate weighting scheme (Figure 13-3) produces a 45 by 3 meter array with uniform illumination. The resulting focused pattern at $2D^2/\lambda$ is shown in Figure 13-17. The second alternate weighting (Figure 13-4) produced a weighted 9 by 9 meter array. This pattern is shown in Figure 13-18. The last weighting used (Figure 13-5) illuminates the entire TA-2 alternate configuration; this pattern is presented in Figure 13-19.

13.3 TA-1 ANTENNA PATTERNS

The present Solar Power Satellite (SPS) TA-1 antenna concept is a cross array of dimension 123^m by 125.6^m . Both the horizontal and the vertical arrays are sectionalized, with waveguide lengths varying from 2.39 to 28.72m. Each section is fed with an amplatron to achieve the space tapering. All sections are made of standard waveguide sections (0.4786m).

Figure 13-20 shows the general configuration of the cross array and the associated notations used. Table 13-1 gives numerical values of various dimensions.

The antenna has an elliptical beam shape at 3 dB level. The beamwidths of the tapered amplitude as compared with those of the uniform illumination are:

	<u>Uniform Illumination</u>	<u>Tapered Illumination</u>
ψ		
0°	0.07°	0.121°
45°	0.07°	0.11°
90°	0.082	0.16°

Thus the tapering of amplitudes appreciably increases the beamwidth. The results also show that the aspect ratio of the beam cross section is a function of the horizontal and the vertical array widths.

The above result is obtained via the following mathematical expressions which are applicable for a general cross-array configuration.

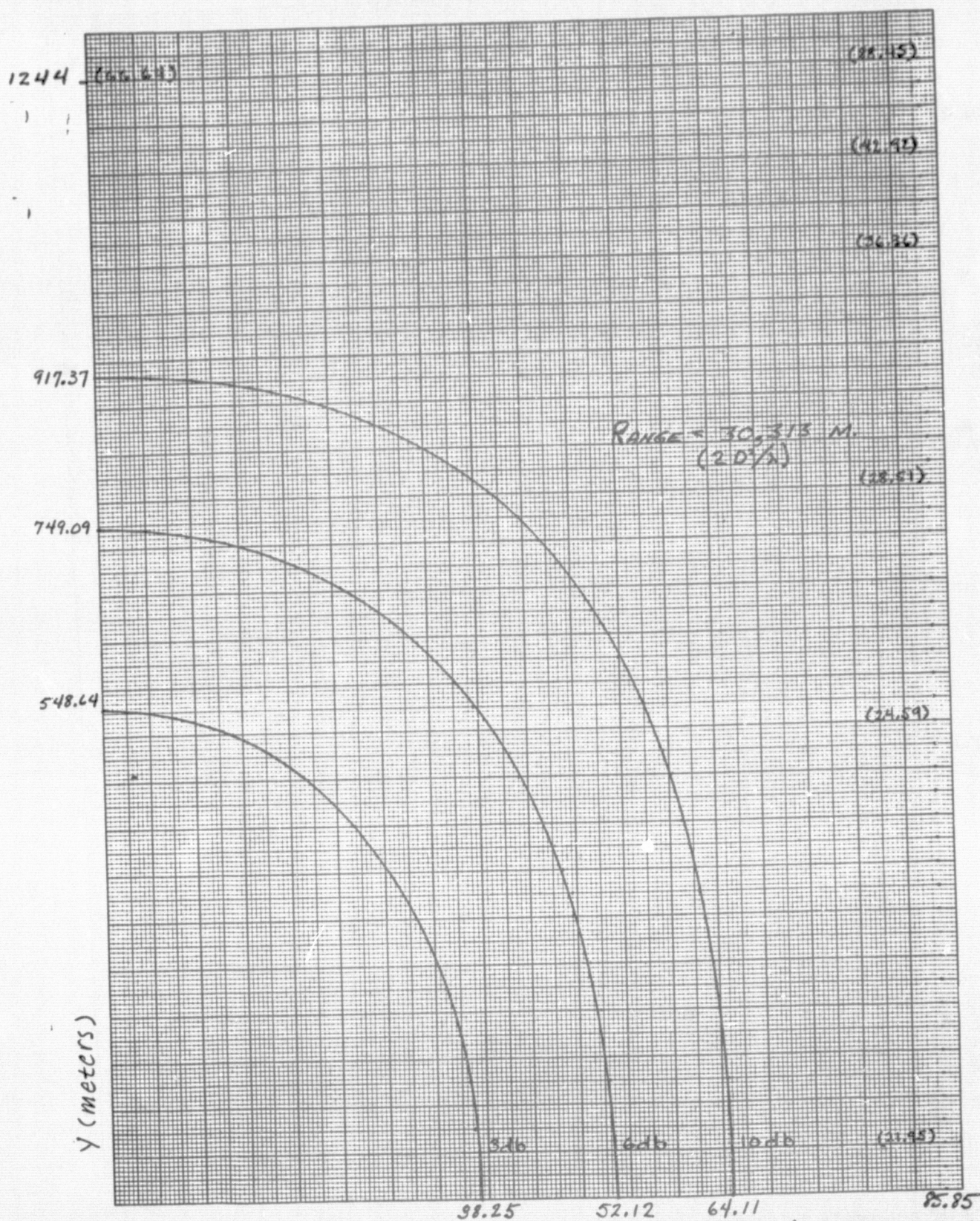


Figure 13-17. Alternate TA-2, Uniform Weighting

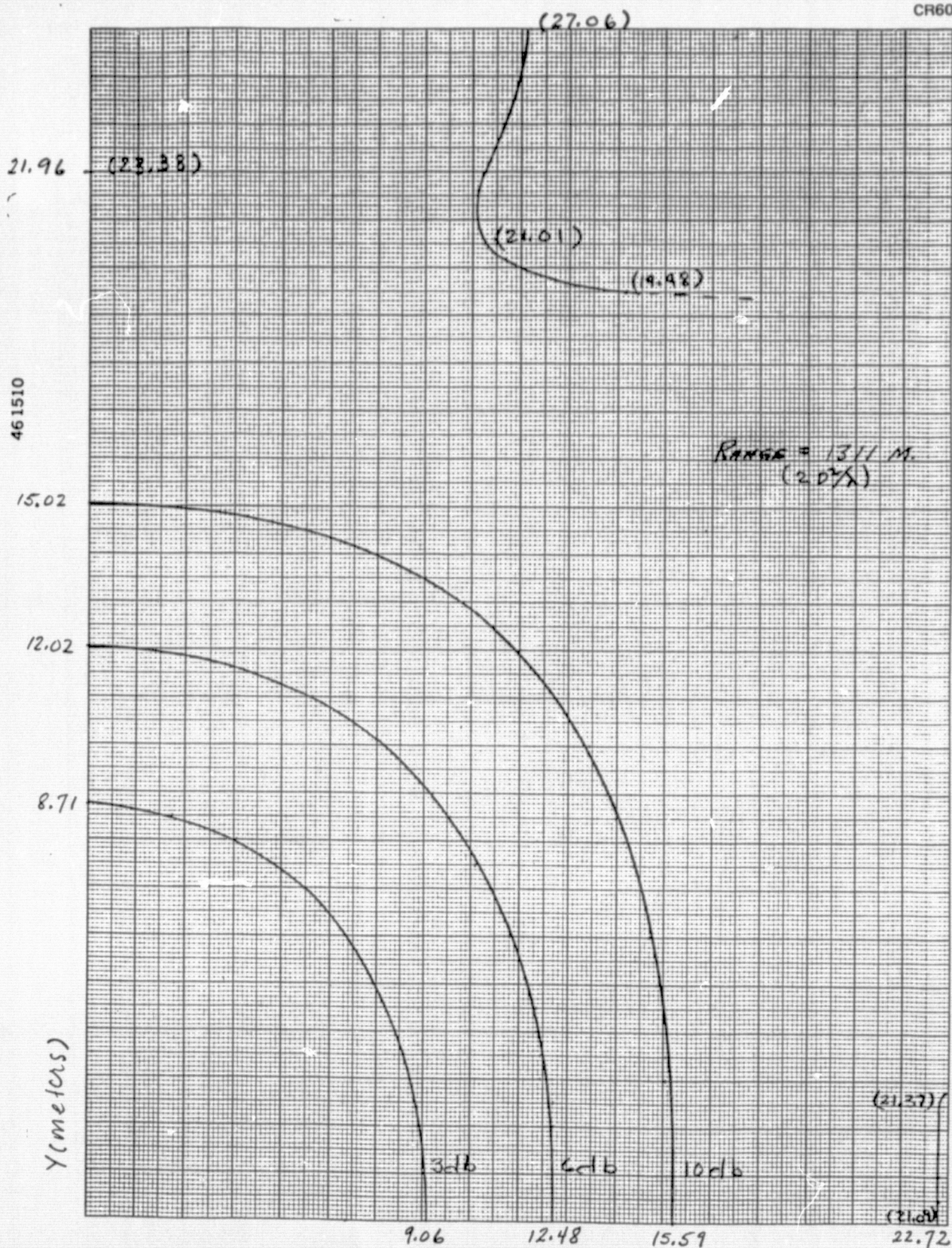


Figure 13-18. Alternate TA-2, 9 x 9 Weighted

TA1

2	2	2	1
2	2	2	1
2	2	2	1

$M.C. = \sqrt{2}$
 $W.L. = \sqrt{2}$
 $W.L. = 1$

CR60

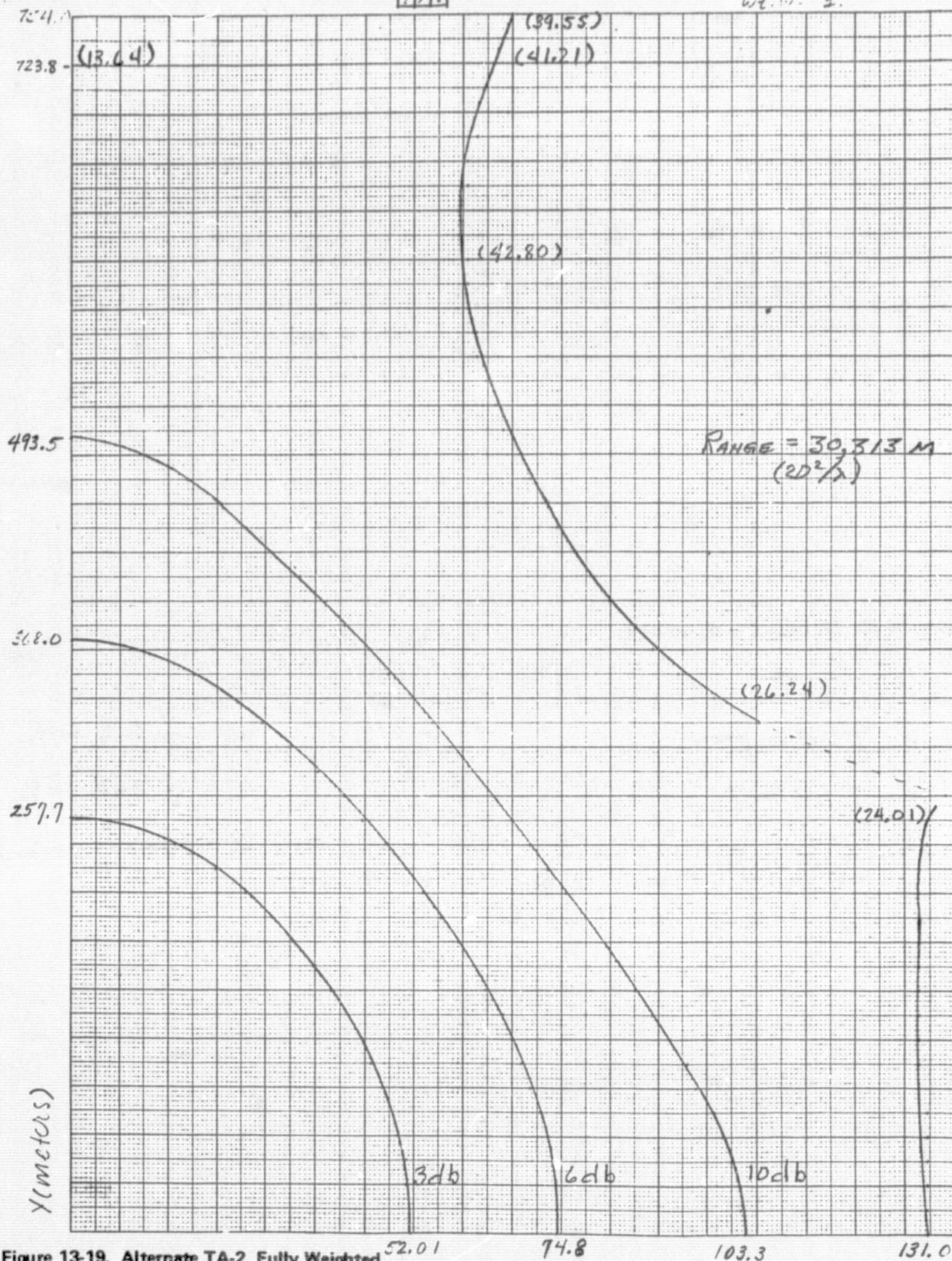


Figure 13-19. Alternate TA-2, Fully Weighted

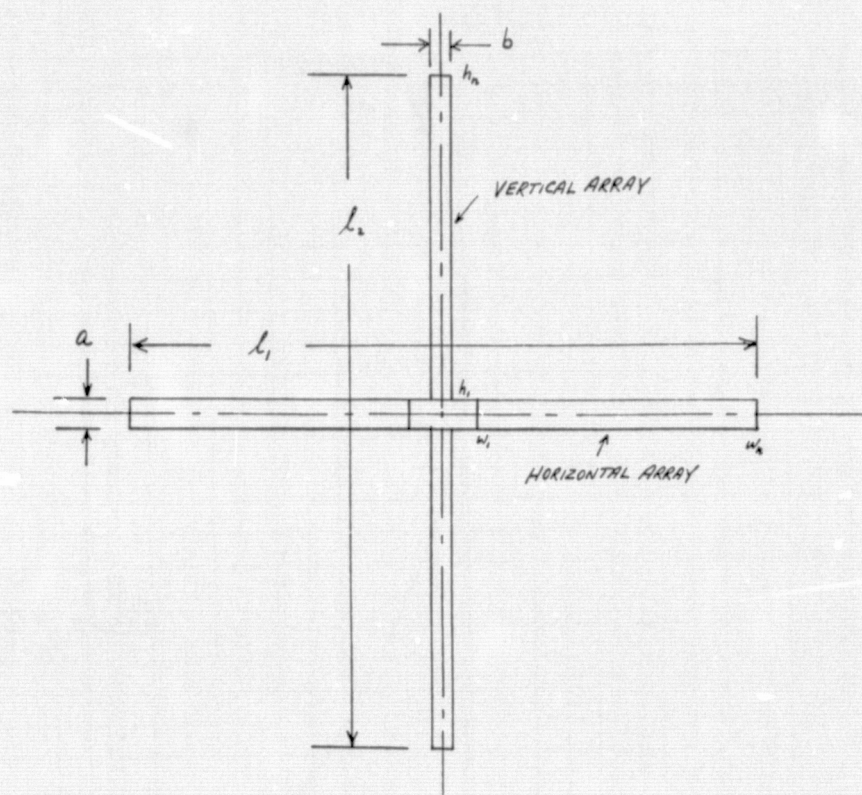


Figure 13-20. TA-1 Cross Array Geometry

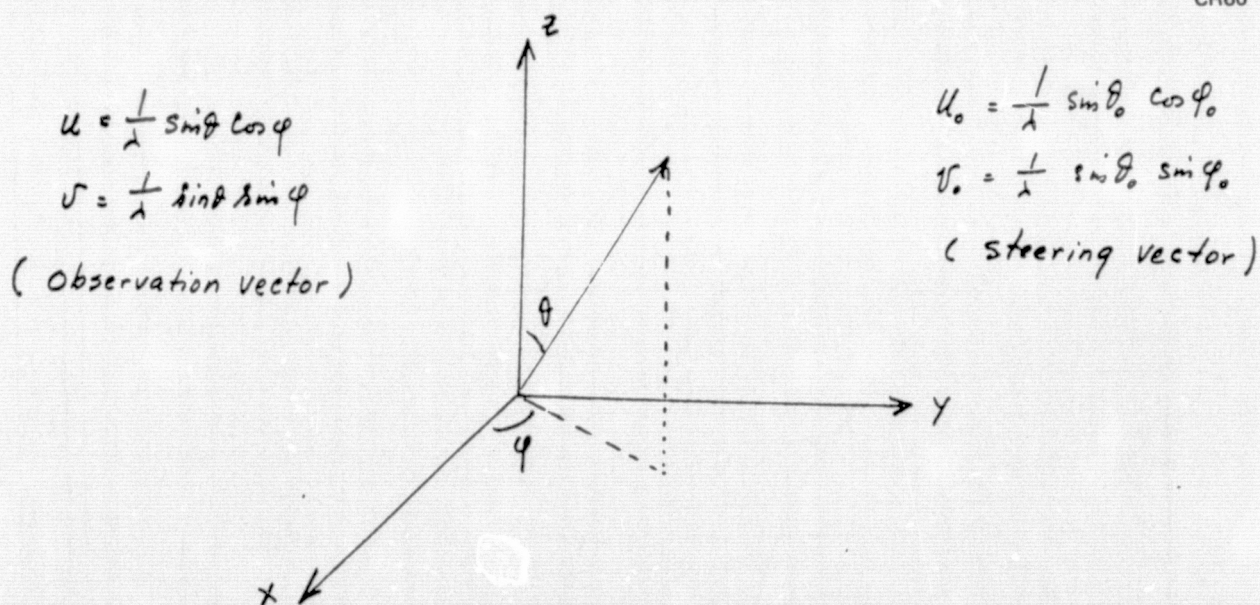


Figure 13-21. Coordinate System

Referring to Figure 13-21, the horizontal array is along the x-axis and the vertical array the y-axis. θ and ϕ are polar angles of the observation vector. Let

$$u = \frac{1}{\lambda} \sin \theta \cos \phi$$

$$v = \frac{1}{\lambda} \sin \theta \sin \phi$$

Table 13-1
CROSS ARRAY DIMENSIONS

	Horizontal Array	Vertical Array
Array Length	123 ^m	125.6 ^m
Array Width	0.199m	0.149m
Waveguide Slots (0.0498 x 0.0995 ^m cross section)	Broad Wall	Side Wall
Standard Waveguide Section	$l = 0.4786\text{m} \cong 4\lambda$ $\lambda = 0.1225\text{m}$ $2W_1 = h_2 - h_1 = 2.39\text{m} = 5l$ $W_2 - W_1 = h_3 - h_2 = 2.87\text{m} = 6l$ $W_3 - W_2 = h_4 - h_3 = 4.79\text{m} = 10l$ $W_4 - W_3 = h_5 - h_4 = 9.57\text{m} = 20l$ $W_5 - W_4 = h_6 - h_5 = 14.36^{\text{m}} = 30l$ $W_6 - W_5 = h_7 - h_6 = 28.72^{\text{m}} = 60l$	
Wavelength		
Waveguide Lengths		
Section Centers	$M_1 = 5.5l$	$N_1 = 2.5l$
	$M_2 = 13.5l$	$N_2 = 8l$
	$M_3 = 28.5l$	$N_3 = 16l$
	$M_4 = 53.5l$	$N_4 = 31l$
	$M_5 = 98.5l$	$N_5 = 56l$
		$N_6 = 101l$

The radiation pattern of a cross-array can in general be written as

$$F(u, v; u_0, v_0) = \frac{a \sin(\pi a v)}{(\pi a v)} I_1(u, u_0) + \frac{b \sin(\pi b u)}{(\pi b u)} I_2(v, v_0)$$

Where I_1 and I_2 are patterns of the horizontal array and the vertical array respectively.

To evaluate the cross-array radiation pattern, expressions for I_1 and I_2 have been derived for various array amplitudes and phases:

A. Broadside beam - uniform amplitude

$$I_1(u) = \frac{\ell_1 \sin(\pi \ell_1 u)}{(\pi \ell_1 u)}$$

$$I_2(v) = \frac{\ell_2 \sin(\pi \ell_2 v)}{(\pi \ell_2 v)} - \frac{a \sin(\pi a v)}{(\pi a v)}$$

B. Broadside beam - tapered amplitude

$$I_1(u) = \frac{1}{\pi u} \sum_{i=1}^n (G_i - G_{i+1}) \sin(2\pi w_i u), \quad G_i = 0 \text{ for } i > n$$

$$I_2(v) = \frac{1}{\pi v} \sum_{i=1}^n (H_i - H_{i+1}) \sin(2\pi h_i v), \quad H_1 = 0, \quad H_i = 0 \text{ for } i > n$$

C. Scanned beam - uniform amplitude, continuous phase

$$I_1(u, u_0) = \frac{\sin 2\pi w_n (u - u_0)}{\pi (u - u_0)}$$

$$I_2(v, v_0) = \frac{1}{\pi (v - v_0)} [\sin 2\pi h_n (v - v_0) - \sin 2\pi h_1 (v - v_0)]$$

D. Scanned beam - tapered amplitude, continuous phase

$$I_1(u, u_0) = \frac{1}{\pi(u-u_0)} \sum_{i=1}^n (G_i - G_{i+1}) \sin(2\pi w_i(u-u_0)), \quad G_i = 0 \text{ for } i > n$$

$$I_2(v, v_0) = \frac{1}{\pi(v-v_0)} \sum_{i=1}^n (H_i - H_{i+1}) \sin(2\pi h_i(v-v_0)), \quad H_i = 0, \quad H_i = 0 \text{ for } i > n$$

E. Scanned beam - tapered amplitude, one phase shifter per amplatron

$$I_1(u, u_0) = \frac{1}{\pi u} \sum_{i=1}^n 2G_i \sin(\pi(w_i - w_{i-1})u) \cos 2\pi \left[\frac{w_i + w_{i-1}}{2} u - M_{i-1}u_0 \right], \quad M_0 = 0$$

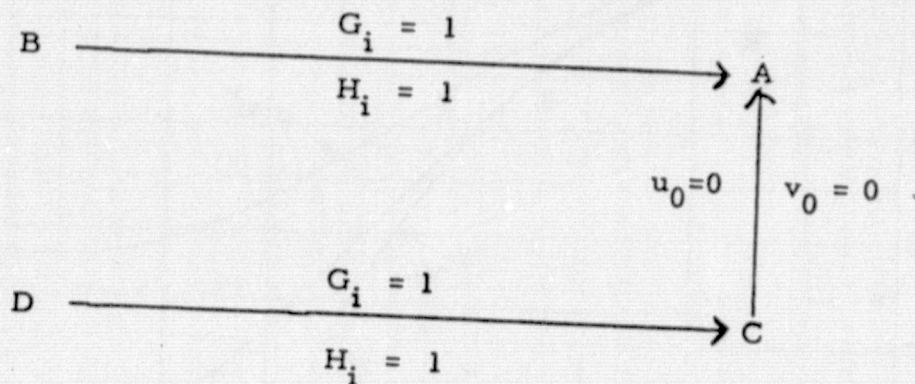
$$I_2(v, v_0) = \frac{1}{\pi v} \sum_{i=2}^n 2H_i \sin(\pi(h_i - h_{i-1})v) \cos 2\pi \left[\frac{h_i + h_{i-1}}{2} u - N_i v_0 \right]$$

F. Scanned beam - tapered amplitude, one phase shifter per 2.39 m waveguide section.

$$I_1(u, u_0) = \frac{1}{\pi u} \left\{ G_1 \sin 2\pi u w_1 + G_2 \left[\sin 2\pi (3u - 2u_0) w_1 - \sin 2\pi (u - 2u_0) w_1 + \sin 2\pi (u w_2 - 4u_0 w_1) \right. \right. \\ \left. - \sin 2\pi (3u - 4u_0) w_1 \right] + G_3 \left[\sin 2\pi (5u - 4u_0) w_1 - \sin 2\pi (u w_2 - 4u_0 w_1) + \sin 2\pi (7u - 6u_0) w_1 \right. \\ \left. - \sin 2\pi (5u - 6u_0) w_1 + \sin 2\pi (u w_3 - 8u_0 w_1) - \sin 2\pi (7u - 8u_0) w_1 \right] \\ + G_4 \left[\sin 2\pi (9u - 8u_0) w_1 - \sin 2\pi (u w_3 - 8u_0 w_1) + \sum_{i=11, 13, 15} 2 \sin 2\pi u w_i \cos(i-1) 2\pi w_i (u - u_0) \right. \\ \left. + \sin 2\pi (u w_4 - 16u_0 w_1) - \sin 2\pi (15u - 16u_0) w_1 \right] \\ + G_5 \left[\sin 2\pi (17u - 16u_0) w_1 - \sin 2\pi (u w_4 - 16u_0 w_1) + \sum_{i=17, \dots, 27} 2 \sin 2\pi u w_i \cos(i-1) 2\pi w_i (u - u_0) \right. \\ \left. + \sin 2\pi (u w_5 - 28u_0 w_1) - \sin 2\pi (27u - 28u_0) w_1 \right] \\ + G_6 \left[\sin 2\pi (29u - 28u_0) w_1 - \sin 2\pi (u w_5 - 28u_0 w_1) + \sum_{i=29, \dots, 49} 2 \sin 2\pi u w_i \cos(i-1) 2\pi w_i (u - u_0) \right. \\ \left. + \sin 2\pi (u w_6 - 50u_0 w_1) - \sin 2\pi (49u - 50u_0) w_1 \right] \left. \right\}$$

$$I_2(v, v_0) = \frac{1}{\pi v} \left\{ H_2 \left[\sin 2\pi (2v - v_0) w_1 + \sin 2\pi v_0 w_1 \right] \right. \\ + H_3 \left[\sin 2\pi (4v - 3v_0) w_1 - \sin 2\pi (2v - 3v_0) w_1 + \sin 2\pi (v h_3 - 5v_0 w_1) - \sin 2\pi (4v - 5v_0) w_1 \right] \\ + H_4 \left[\sin 2\pi (6v - 5v_0) w_1 - \sin 2\pi (v h_3 - 5v_0 w_1) + \sin 2\pi (8v - 7v_0) w_1 - \sin 2\pi (6v - 7v_0) w_1 \right. \\ \left. + \sin 2\pi (v h_4 - 9v_0 w_1) - \sin 2\pi (8v - 9v_0) w_1 \right] \\ + H_5 \left[\sin 2\pi (10v - 9v_0) w_1 - \sin 2\pi (v h_4 - 9v_0 w_1) + \sum_{i=12, 14, 16} 2 \sin 2\pi v w_i \cos(i-1) 2\pi w_i (v - v_0) \right. \\ \left. + \sin 2\pi (v h_5 - 17v_0 w_1) - \sin 2\pi (16v - 17v_0) w_1 \right] \\ + H_6 \left[\sin 2\pi (18v - 17v_0) w_1 - \sin 2\pi (v h_5 - 17v_0 w_1) + \sum_{i=20, \dots, 28} 2 \sin 2\pi v w_i \cos(i-1) 2\pi w_i (v - v_0) \right. \\ \left. + \sin 2\pi (v h_6 - 29v_0 w_1) - \sin 2\pi (28v - 29v_0) w_1 \right] \\ + H_7 \left[\sin 2\pi (30v - 29v_0) w_1 - \sin 2\pi (v h_6 - 29v_0 w_1) + \sum_{i=32, \dots, 50} 2 \sin 2\pi v w_i \cos(i-1) 2\pi w_i (v - v_0) \right. \\ \left. + \sin 2\pi (v h_7 - 51v_0 w_1) - \sin 2\pi (50v - 51v_0) w_1 \right] \left. \right\}$$

By applying special conditions, various expressions can be reduced. For example:



Various computer programs were generated; thus, an analytical tool exists for evaluating the cross array performance. Several results are included here. Figures 13-22 and 13-23 patterns for the uniform illumination and the tapered illumination respectively. Since the space tapering is achieved by feeding the same amplatron power to different length waveguides, we assume

$$G_i \propto \frac{1}{W_i - W_{i-1}}$$

$$H_i \propto \frac{1}{h_i - h_{i-1}}$$

Figure 13-24 shows a 3D plot of the tapered illumination pattern.

13.4 FURTHER ANALYSIS OF TA-1 ANTENNA PATTERNS

The evaluation of the TA-1 cross array radiation pattern was discussed in Section 13-3. This section summarizes additional results obtained for the baseline antenna configuration.

Considerations were given to various possible testing modes for the cross array antenna used as a versatile test article. They include:

Case 1 — Both the horizontal and the vertical arrays are in operation; each array consists of two waveguide array radiators (2H + 2V).

Case 2 — Both the horizontal and the vertical array are in operation, but only one of the waveguide array radiators in each array is radiating (H+V).

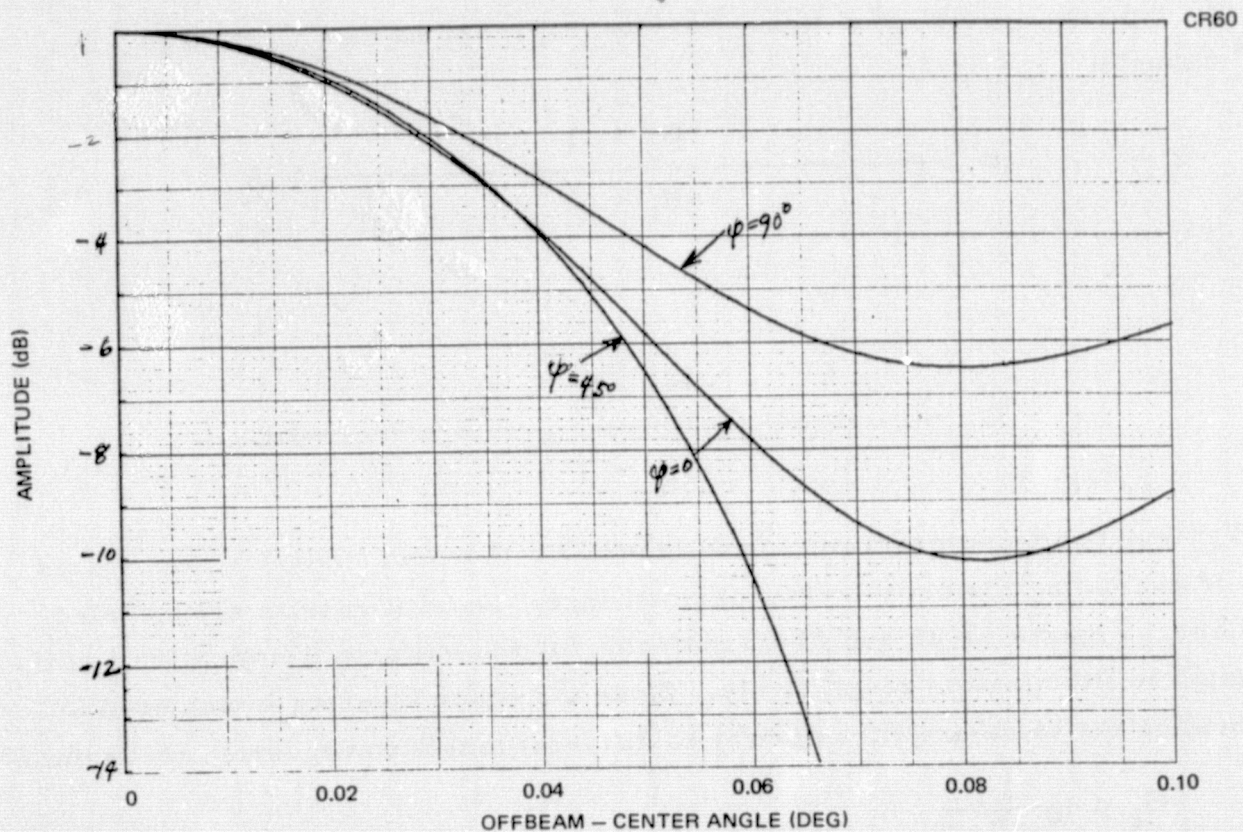


Figure 13-22. TA-1 Patterns for Uniform Illumination

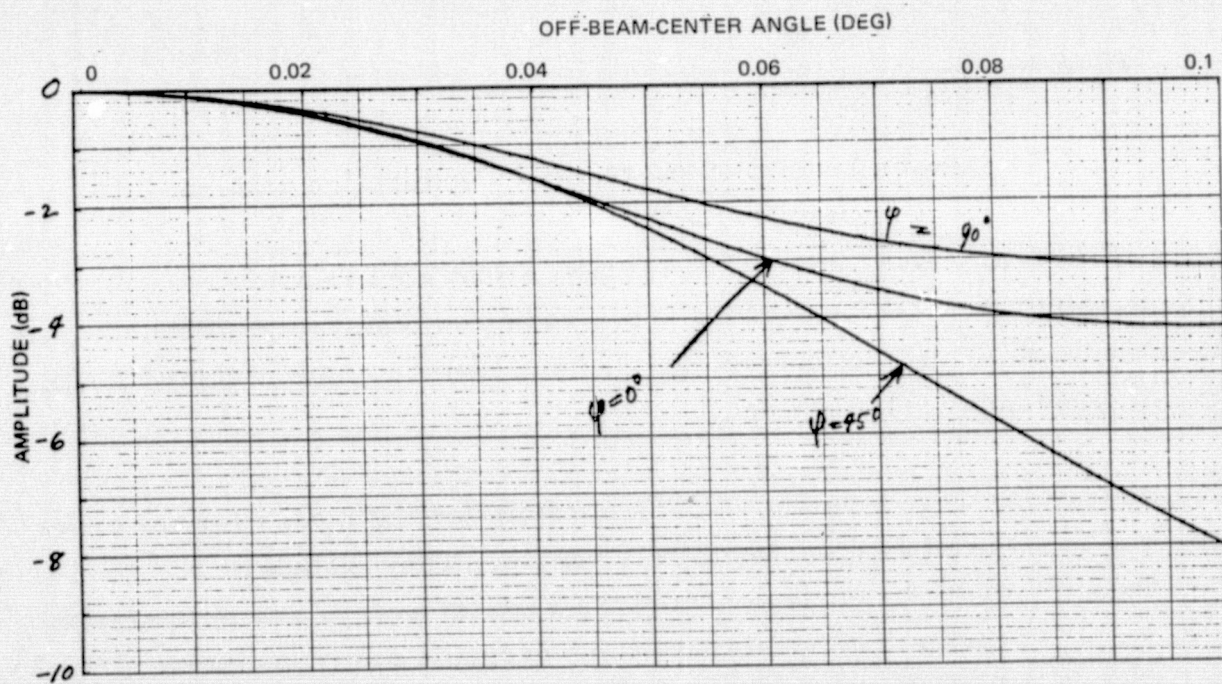


Figure 13-23. TA-1 Patterns for Tapered Illumination

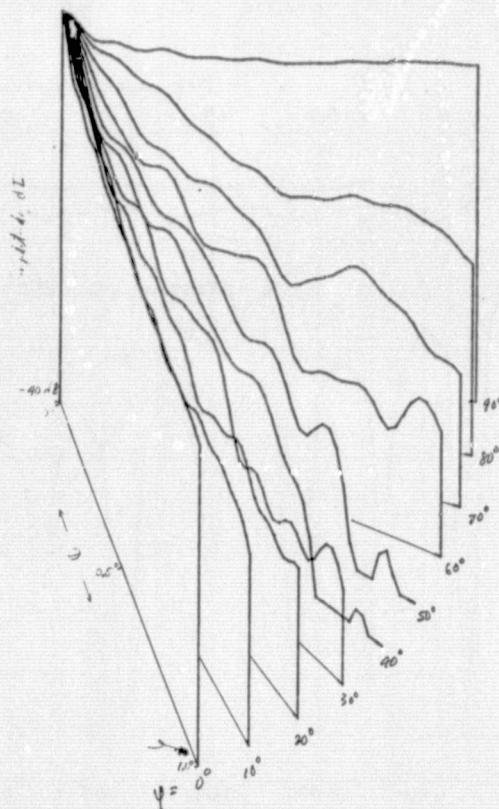


Figure 13-24. 3D Pattern for Tapered Illumination (TA-1)

Case 3 — Only the horizontal array is operational—two waveguides (2H).

Case 4 — Only the vertical array operational—two waveguides (2V).

Case 5 — Only the horizontal array operational—one waveguide (H).

Case 6 — Only the vertical array operational—one waveguide (V).

Preliminary calculations of these antenna patterns were performed. The results are summarized below:

Case	Uniform Illumination	Tapered Illumination
1 (2H + 2V)	0.07°	0.12°
2 (H + V)	0.07°	0.12°
3 (2H)	0.05°	0.078°
4 (2V)	0.05°	0.079°
5 (H)	0.05°	0.078°
6 (V)	0.05°	0.079°

Thus the cross array has an appreciably different main beam characteristic as compared to each individual single array. Pattern plots are shown in Figure 13-25 through 13-31. Results for the Case 2H + 2V were presented in Section 13-3. Case 2 (H+V) and Case 5 (H) are most typical of TA-1 operation.

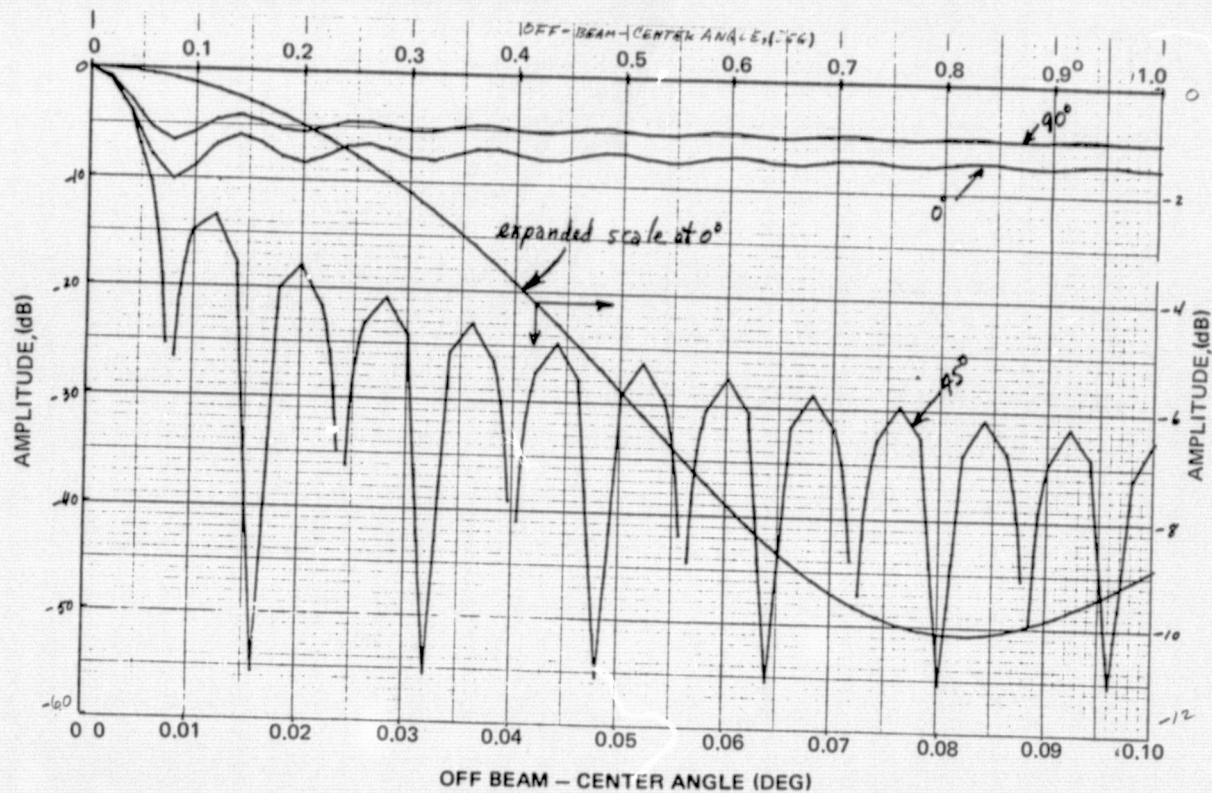


Figure 13-25. TA-1 Half Width (H+V) Cross Array Pattern - Uniform

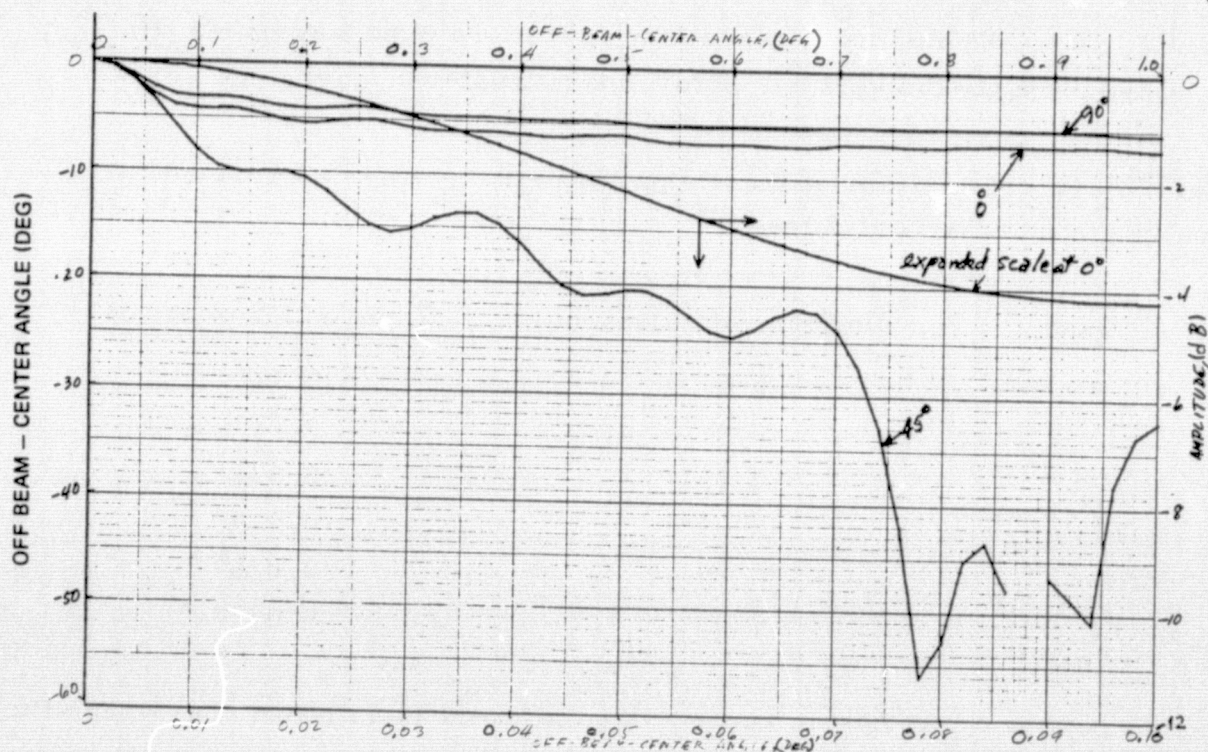


Figure 13-26. TA-1 Half Width (H+V) Cross Array Pattern - Tapered

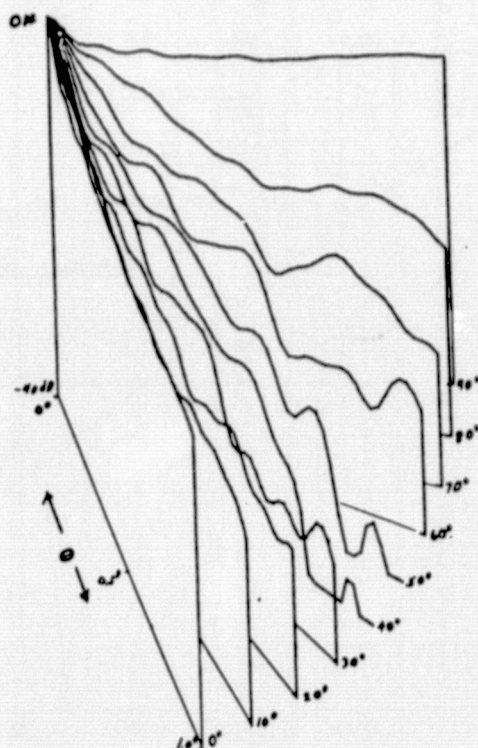


Figure 13-27. TA-1 Half Width Cross Array (H+V) 3D Pattern — Tapered Illumination

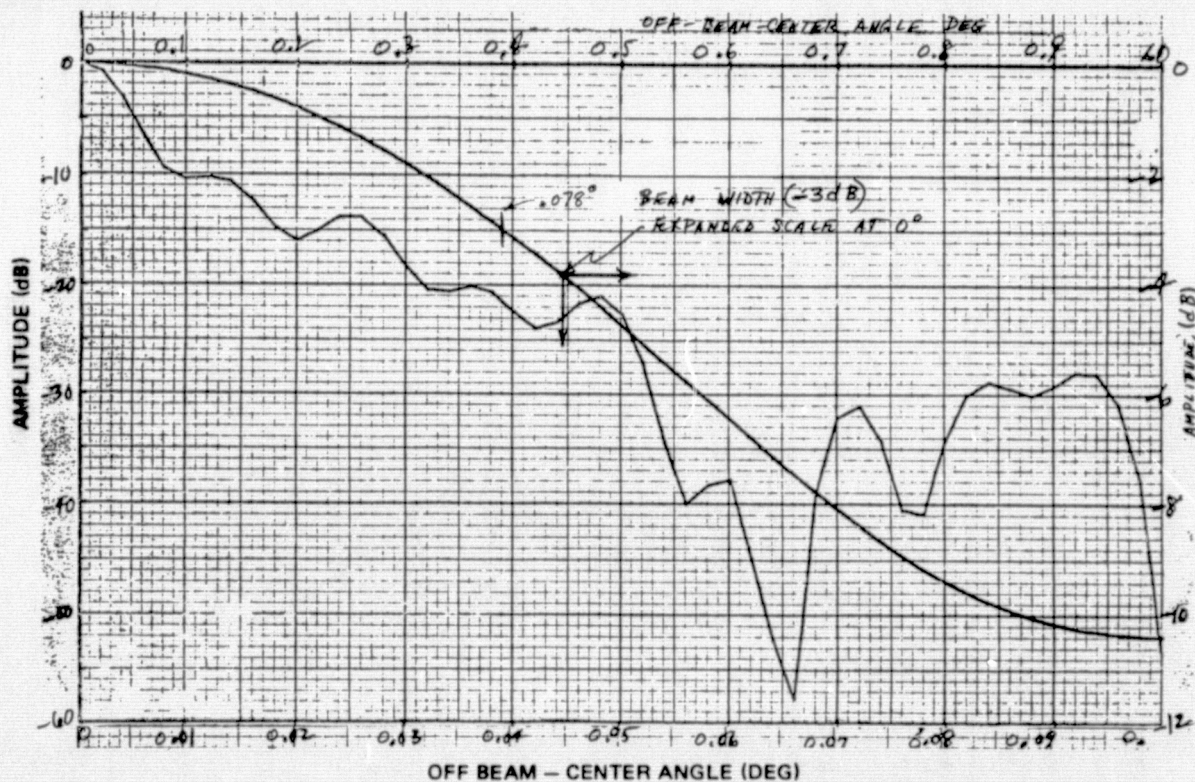


Figure 13-28. TA-1 Horizontal Array Pattern (2H) — Tapered Illumination

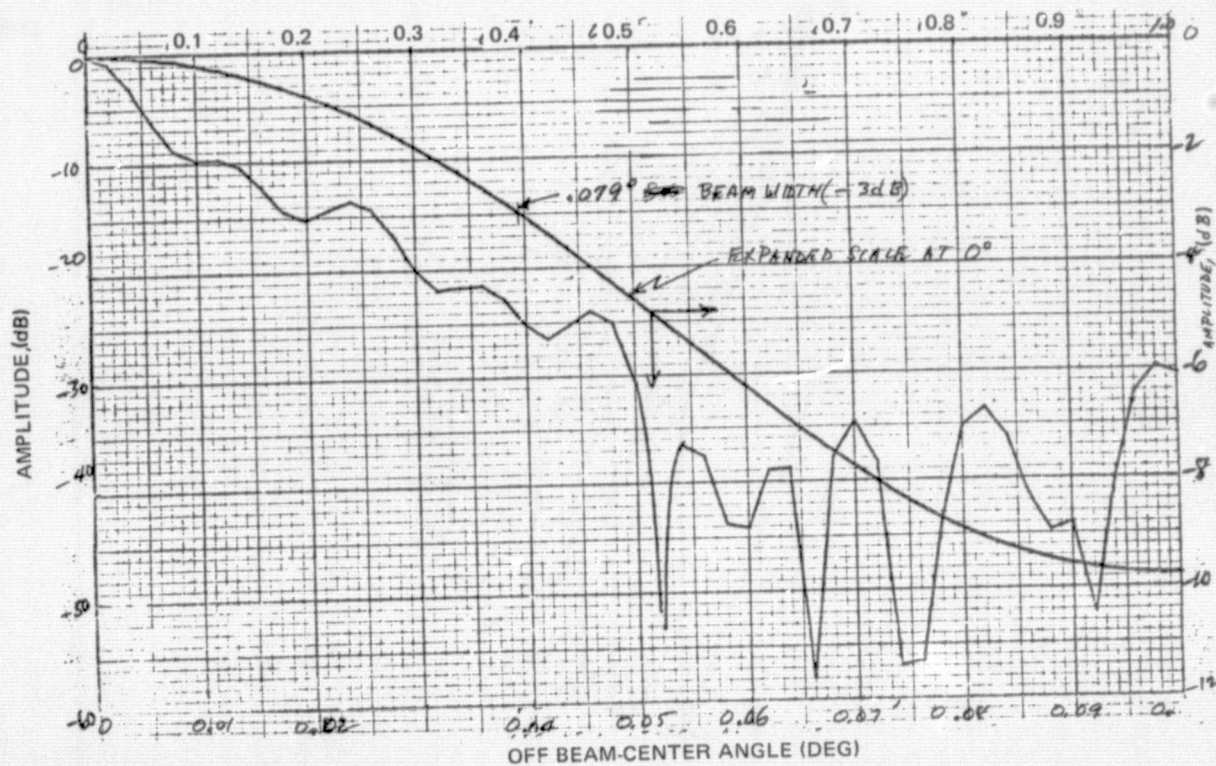


Figure 13-29. TA-1 Vertical Array Pattern (2V) - Tapered Illumination

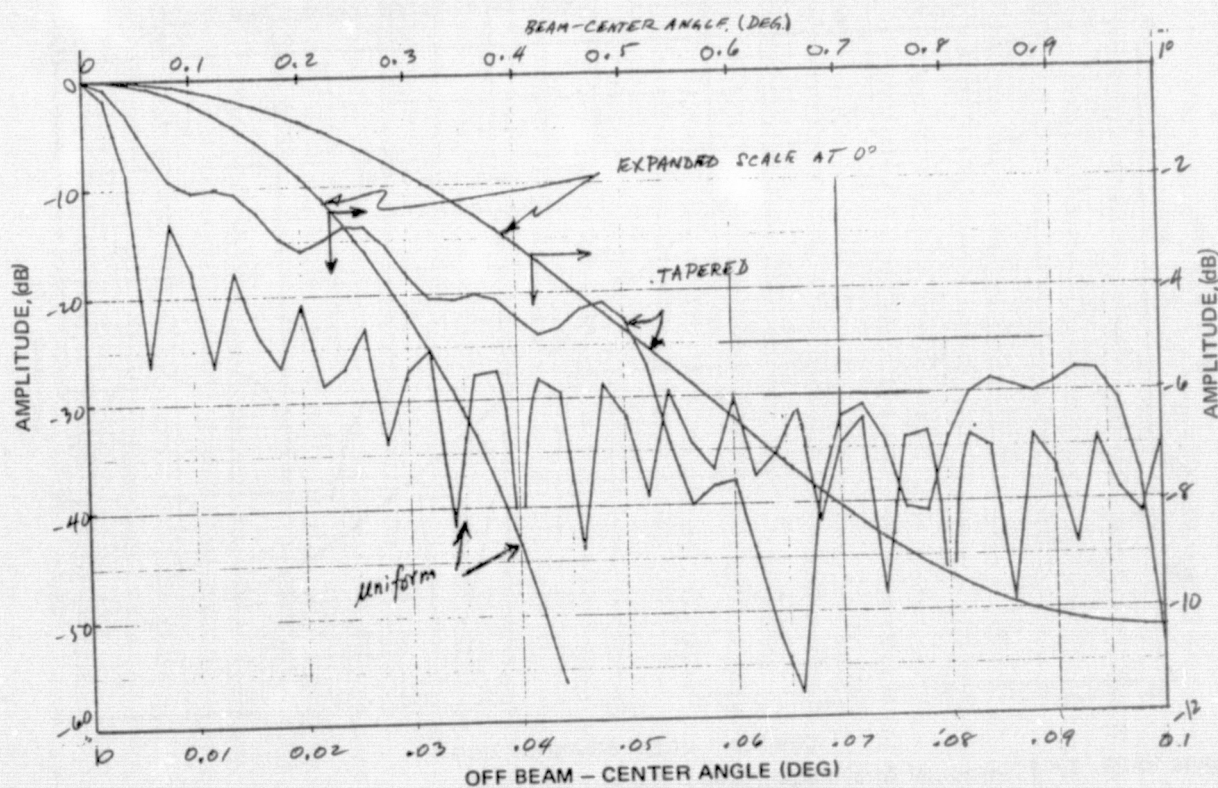


Figure 13-30. TA-1 Horizontal Array Pattern (H) - Uniform and Tapered

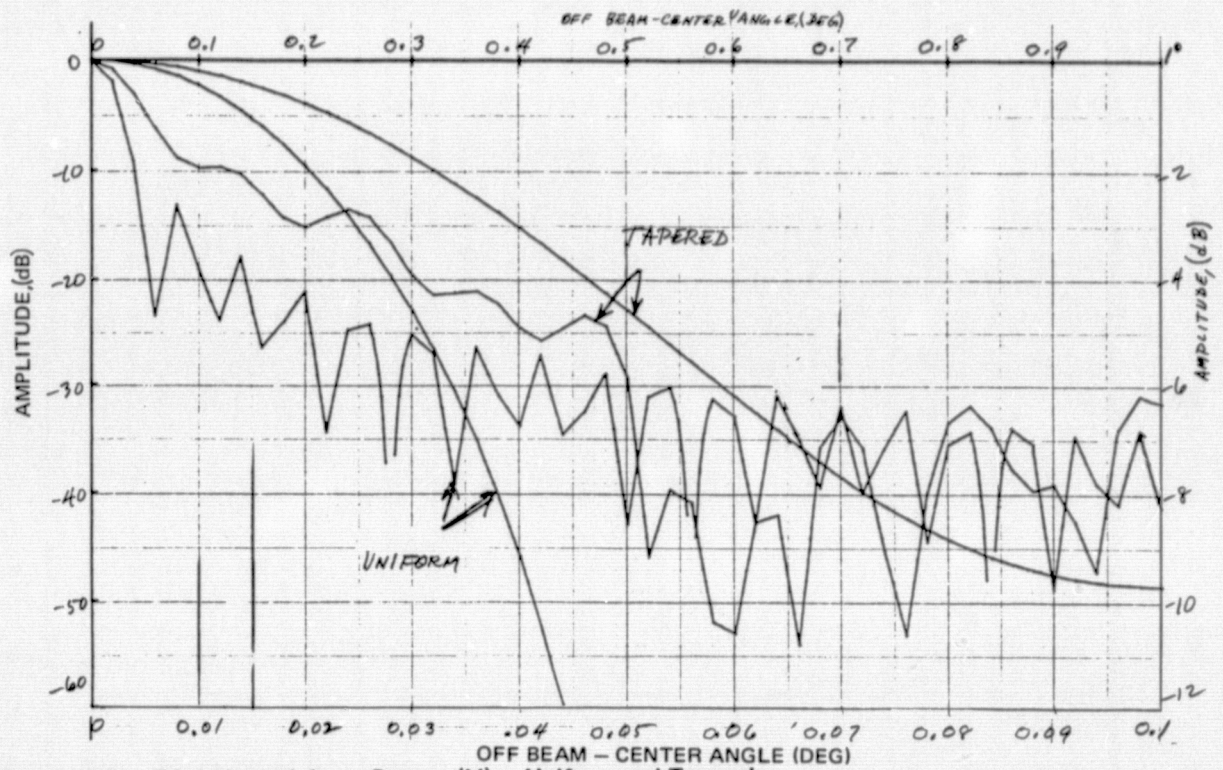


Figure 13-31. TA-1 Vertical Array Pattern (V) - Uniform and Tapered

Section 14

SPS TEST ARTICLE MISSION HARDWARE

14.1 SUBARRAY SIZING CONSTRAINTS

A planar array antenna consists of multiple antennas with controlled-phase relationship between antennas for steering instead of a single antenna, which is mechanically moved for steering. This antenna configuration is optimally suited for the SPS application; however, the constraints of a multiple-antenna system must be considered if maximum efficiency and minimum spurious radiation are to be retained over the required system operating conditions.

The SPS antenna is composed of multiple-slotted waveguide radiators connected in subarrays. These subarrays are individually phase-controlled to electronically steer the composite radiated pattern as desired. Each radiating slot produces an element radiation pattern of approximately 60 degrees beamwidth. The subarray combines the element patterns into a subarray radiation pattern whose bandwidth, in radians, is equal to the wavelength divided by the subarray length. Likewise, the array combines the subarray radiation patterns into a composite pattern. Figure 14-1 shows the composite pattern for a 100-meter, unweighted array composed of 10-meter subarrays. Pattern convolution produces a comb of beams spaced a subarray beamwidth apart. Fortunately, in the nonsteered condition shown, these convolutions, or "grating lobes," lie in the subarray pattern nulls and are completely suppressed.

When the array is steered, the center of the composite pattern separates from the subarray pattern center by the steering angle. A 3-milliradian (0.172°) steering angle with 10-meter subarrays is shown in Figure 14-2. Under these conditions, the grating lobes no longer lie on the nulls of the subarray pattern. Consequently, the energy transmitted in the mainlobe decreases, as shown, and this lost energy is primarily radiated in the two major grating lobes, which are incompletely suppressed.

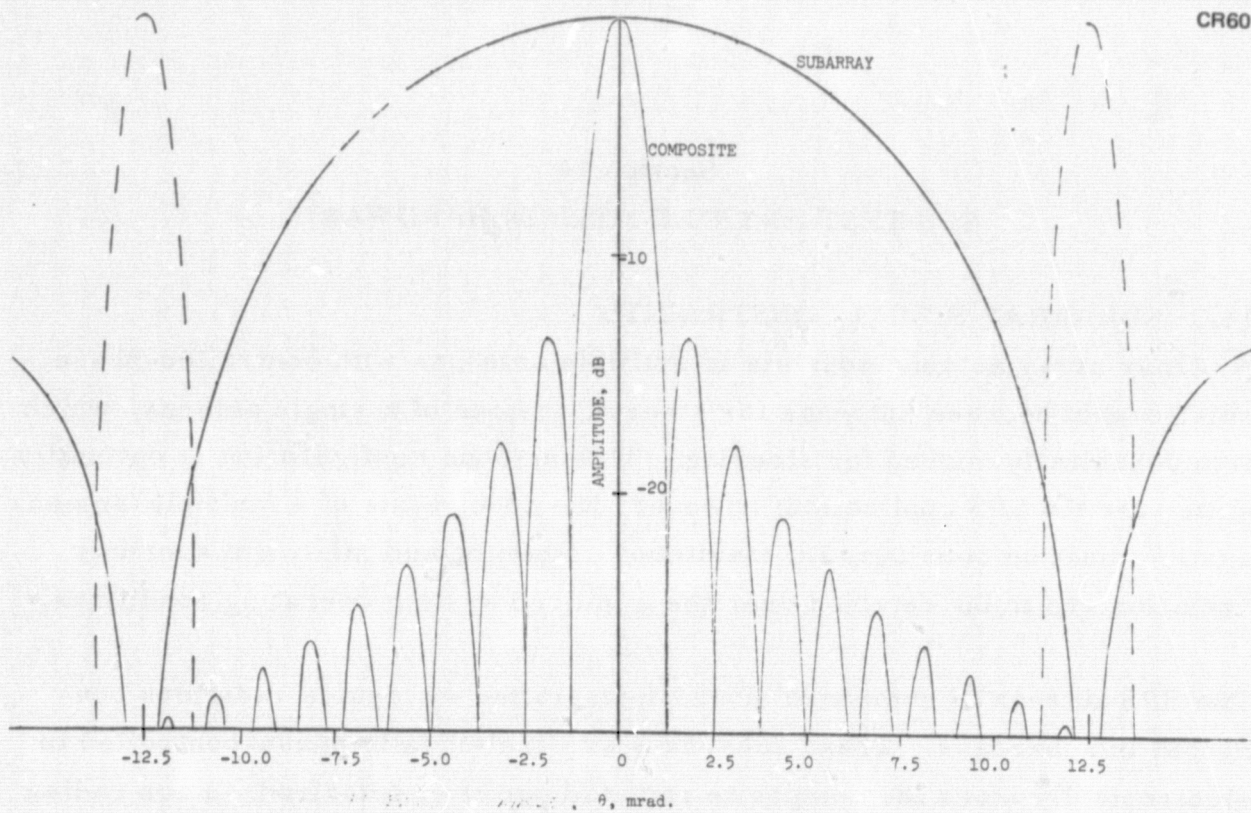


Figure 14-1. Array and Subarray Antenna Pattern (Nonsteered)

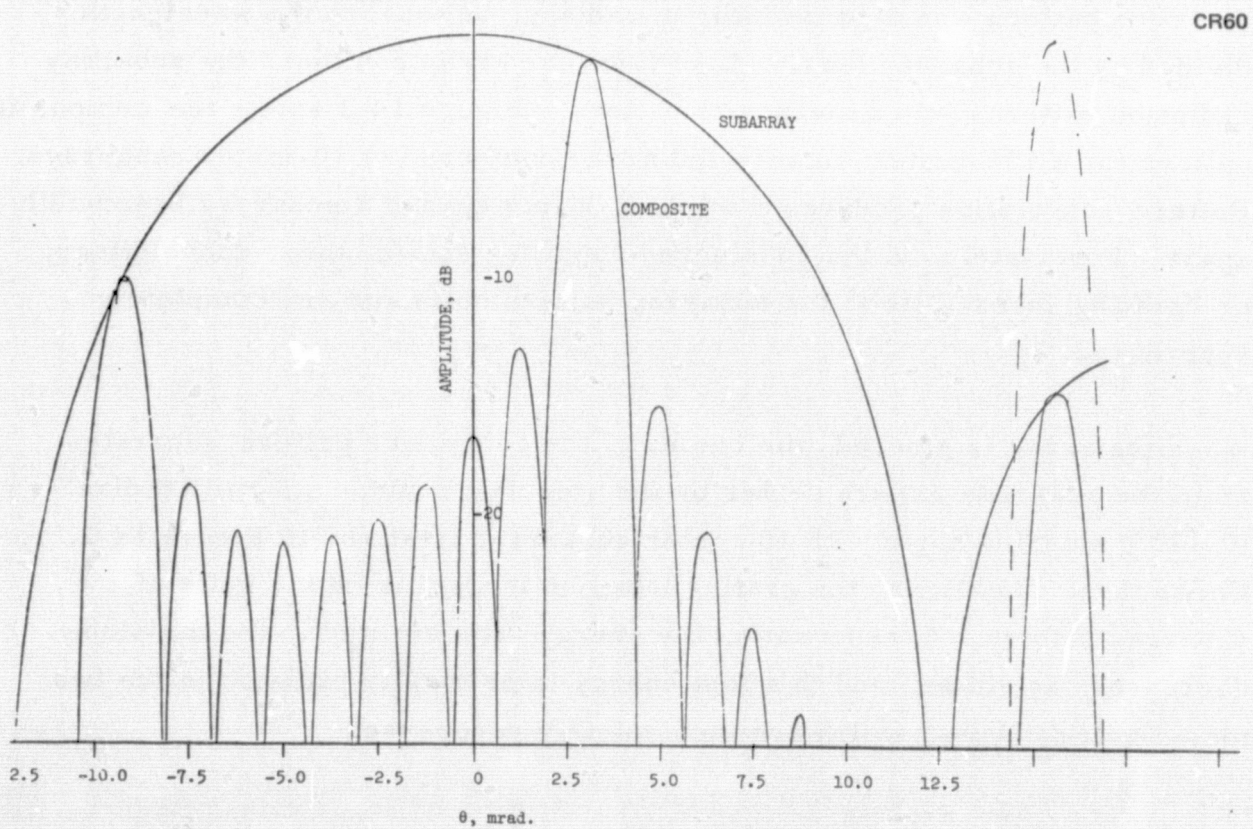


Figure 14-2. Antenna and Subarray Antenna Pattern (Steered)

The magnitude of these two major grating lobes under a 3-milliradian steering condition as a function of subarray size is shown in Figure 14-3. It should be noted that amplitude of these convolutions is independent of the number of subarray involved (total antenna aperture). Figure 14-4 shows that if a 3 milliradian worst-case pilot pulse steering command is anticipated, the phase-controlled subarray should be no larger than 3 by 3 meters if grating lobe amplitude is maintained less than minus 20 db. This analysis assumes no mechanical distortion in the array, which would appear as a steering error to part of the array and add to the grating lobe problem.

Separation of composite and subarray beam centers are also produced by frequency steering of the frequency-sensitive slotted radiators that form the subarray. Consequently, final resolution of the subarray size, thus, the quantity of phase-controlled subarrays required, will depend upon accurate determination of (1) three-sigma mechanical steering error, (2) absolute frequency stability and accuracy of the SPS frequency source, and (3) the uncompensatable variations in resonant frequency of the slotted radiators due to thermal variations encountered in space and power weighting the aperture.

It is therefore, concluded that, in tests directed at development of MPTS antennas, beam mapping should be carried out over polar angles that will include the first grating lobe.

14.2 PILOT RECEIVER CONSIDERATIONS

Precision phase information across the SPS array aperture is required to maintain aperture focus on the receiving antenna. This is accomplished by transmitting a pilot signal from the center of the rectenna and sensing the required relative phase shift required across the SPS antenna array to produce focusing on the pilot transmitter location.

Since the SPS antenna array is operating at very large signal levels, the pilot signal must be adequately offset in frequency from the SPS to permit interference-free reception of the weak pilot signal in the presence of the SPS radiated power. Consequently, the sensed phase difference must be corrected by the frequency ratio f_s/f_p (SPS frequency/pilot frequency).

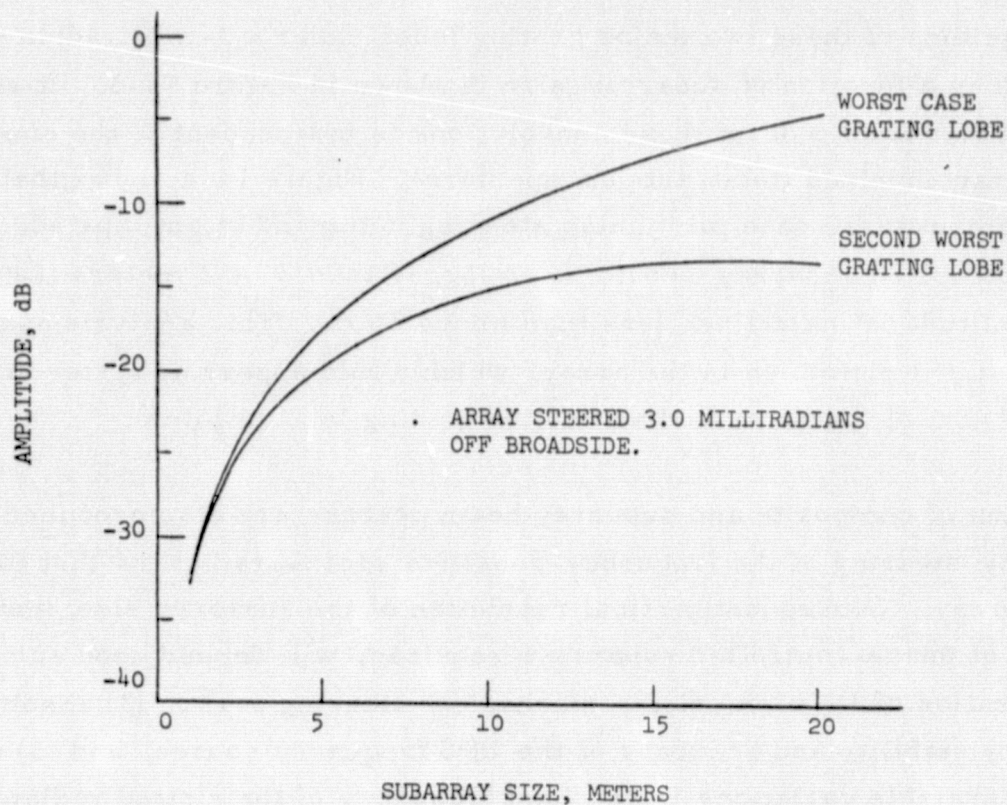


Figure 14-3. Grating Lobes vs Subarray Size

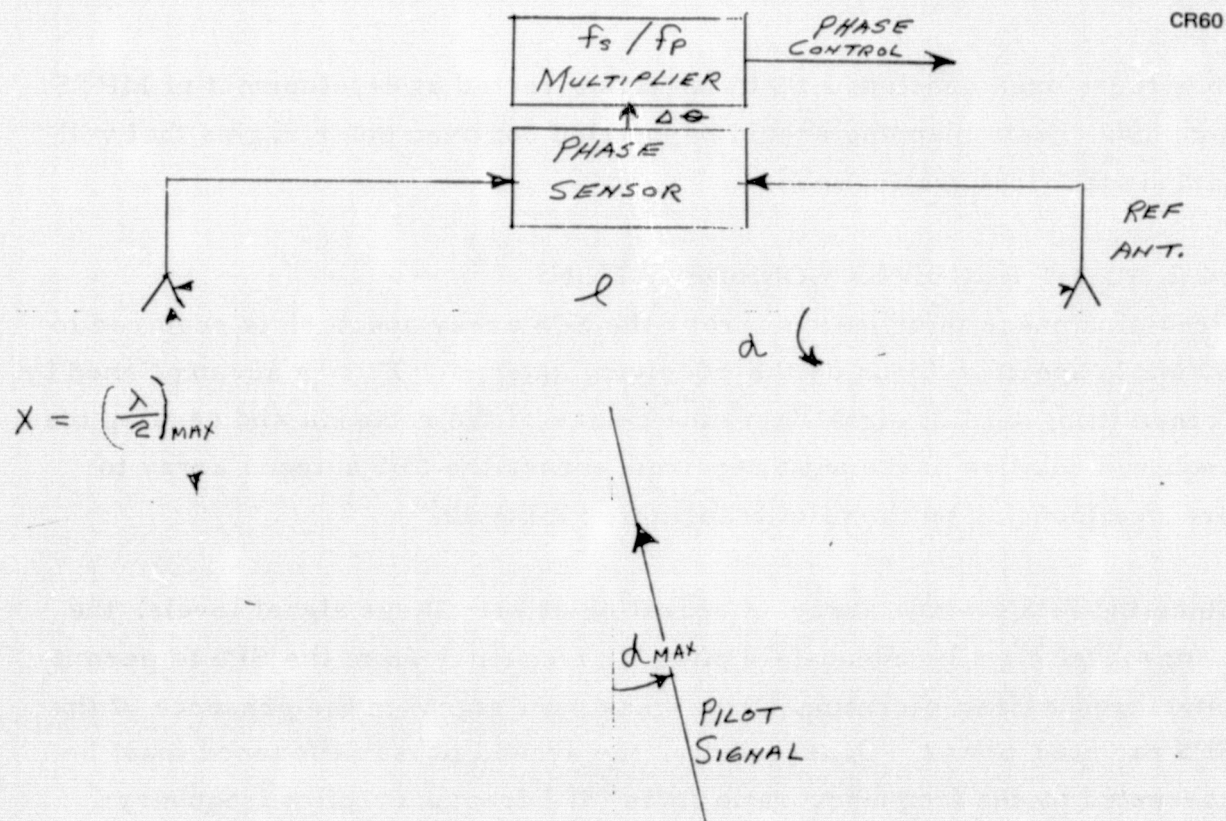


Figure 14-4. Pilot Signal Phase Sensing

This frequency correction requirement imposes constraints on the maximum spacing between pilot receiving antennas.

The phase-sensing configuration constraints are shown graphically in Figure 14-5. Under the worst-case steering requirements of $\pm \alpha_{MAX}$, the path length difference "X" must not exceed one-half wavelength of the pilot signal. A path length difference of +0.75 and a path length difference of -0.25 will both produce a -90 degree phase shift instead of +270 and -90. Consequently, if the f_s/f_p ratio were 1.2, the phase correction would be $1.2 (-90) = -108$ degrees instead of $1.2 (270) = +324$ or -36 degrees. If the maximum path length exceeds a wavelength, the resulting phase error is even more dramatic.

To avoid these erroneous phase control signals, the maximum spacing between phase sensor antennas is as follows:

$$\ell = \frac{\lambda}{2 \tan \alpha_{MAX}}$$

where

ℓ = maximum antenna spacing

λ = pilot signal wavelength

α = maximum angle error due to both mechanical steering and array deformity

If the worst-case angle error can be held to 3 milliradians, the separation between antenna must be less than 20 meters. Conversely, a 6-milliradian (0.34°) composite error will require 10-meter, or less, spacing.

Since the above constraints prohibit phase comparison with a common reference antenna, the delta phase shift between adjacent antennas must be summed with the phase shift requirements ascertained for the adjacent inner phase control zone.

The allowable spacing for pilot antennas exceeds that imposed on the subarray size. However, to minimize electronics, it is recommended that the phase-sensing circuits proportionally control multiple subarrays.

14.3 FREQUENCY SENSITIVITY

The electromagnetic wavelength in a hollow-pipe waveguide, λ_g is related to the cutoff wavelength in the guide λ_c and the free-space wavelength λ by the formula

$$\lambda_g = \frac{\lambda}{\sqrt{1 - \left(\frac{\lambda}{\lambda_c}\right)^2}} \quad (14-1)$$

In rectangular waveguide for any mode of transmission, the cutoff wavelength λ_c is given in terms of the guide dimensions a and b by

$$\lambda_c = \frac{2}{\sqrt{\left(\frac{m}{a}\right)^2 + \left(\frac{n}{b}\right)^2}} \quad (14-2)$$

In this formula, m and n are the subscripts denoting the particular mode under consideration (e. g., $TE_{m,n}$). These two subscripts are integers that indicate the number of half-period variations in transverse field intensity along the x and y dimensions of the guide respectively (see Figure 14-5). For example, the dominant mode in rectangular waveguide is the $TE_{1,0}$ mode, indicating that the wave is of the transverse electric type, that there is a single half-wave variation of transverse field along the x -axis, and that there is no variation of transverse field along the y -axis.

Equation 14-2 holds for either TE or TM modes of transmission. Thus, for the dominant mode ($TE_{1,0}$), we find

$$\lambda_c = 2a = 2(0.0995) = 0.199\text{m} \quad (14-3)$$

At midband $\lambda = \lambda_0 = 0.1225\text{m}$

For radiating slots spaced by the distance d , the beam scan angle is given by

$$\sin \theta = \frac{\lambda}{\lambda_g} - \frac{\lambda}{2d} \quad (14-4)$$

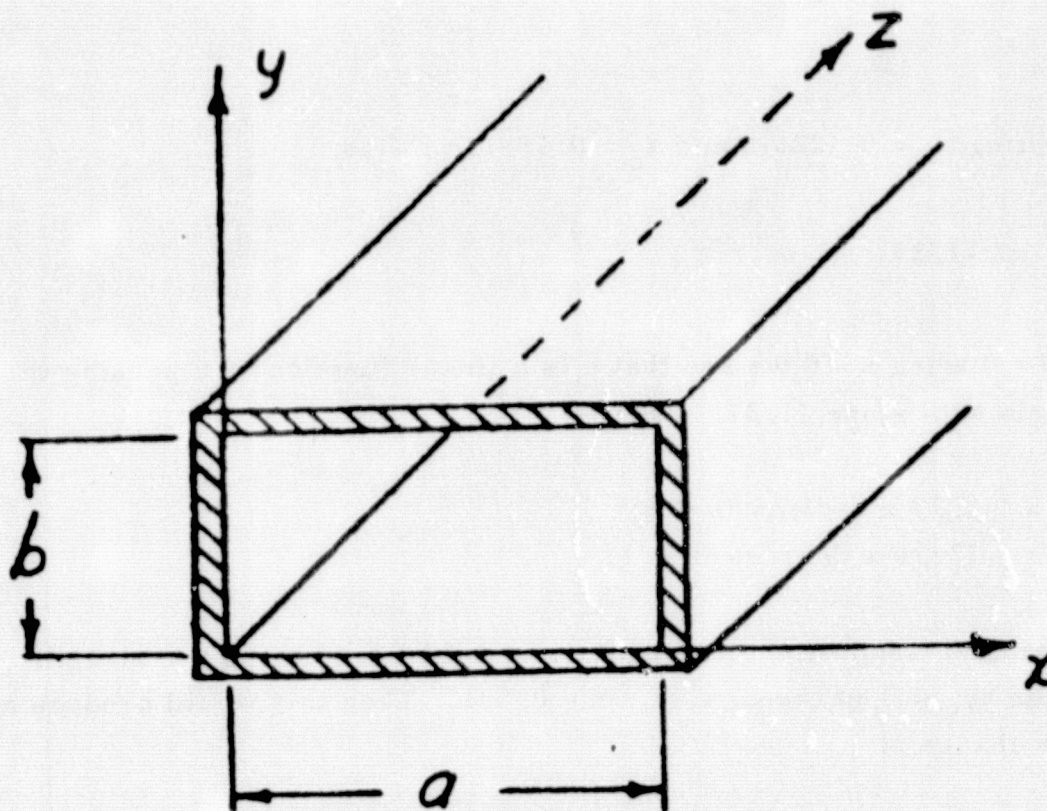


Figure 14-5. Notation Used with Rectangular Waveguides

Thus if we make $d = \lambda_{go}/(2)$, θ will be zero at midband and at any frequency the beam scan angle will given by

$$\sin \theta = \frac{\lambda}{\lambda_g} - \frac{\lambda}{\lambda_{go}} \quad (14-5)$$

Using equation 14-1 in equation 14-5 yields

$$\sin \theta = \sqrt{1 - \left(\frac{\lambda}{\lambda_c}\right)^2} - \frac{\lambda}{\lambda_o} \sqrt{1 - \left(\frac{\lambda_o}{\lambda_c}\right)^2} \quad (14-6)$$

which can be written

$$\sin \theta = \sqrt{1 - \left(\frac{\lambda}{\lambda_o}\right)^2 \left(\frac{\lambda_o}{\lambda_c}\right)^2} - \frac{\lambda}{\lambda_o} \sqrt{1 - \left(\frac{\lambda_o}{\lambda_c}\right)^2} \quad (14-7)$$

Now let

$$\frac{\lambda_o}{\lambda_c} = k < 1 \quad (14-8)$$

Substituting $\lambda_o = 0.1225\text{m}$ and $\lambda_c = 0.199\text{m}$ we find,

$$\frac{d\theta}{dx} = -1.27 \quad (x \ll 1)$$

Since $\theta = 0$ when $x = 0$ we see that θ is a linear function of x passing through the origin with slope -1.27 . That is

$$\begin{aligned}\theta &= -1.27 x \text{ radians } (x \ll 1) \\ \theta &= -72.76 x \text{ degrees } (x \ll 1)\end{aligned}$$

For example, suppose the frequency instability causes the wavelength to change by $+0.1$ percent, i. e., $x = 0.001$. Then this would produce an angular change of

$$\theta = -0.073 \text{ degrees}$$

The fractional change in wavelength x may also be interpreted as a fractional change in frequency since

$$x = \frac{\lambda}{\lambda_o} - 1 = \frac{f_o}{f} - 1 = \frac{f_o - f}{f} \quad (14-16)$$

For small values of x this has almost the same absolute magnitude as

$$y = \frac{f - f_o}{f_o} \quad (14-17)$$

Thus we may say that the 0.073 degree change was produced by a 0.1 percent change in frequency (2.45 MHz). Since the relationship is linear, we find that the frequency sensitivity is approximately 0.03 degrees (0.52 milliradians) per MHz.

Since this subarray tuning error, as well as the steering angle requirement, dictate the maximum allowable subarray size to maintain high efficiency and

Let the fractional wavelength error be x

$$x = \frac{\lambda - \lambda_0}{\lambda_0} = \frac{\lambda}{\lambda_0} - 1 \quad (14-9)$$

Then

$$\sin \theta = \sqrt{1 - (1+x)^2 k^2} - (1+x) \sqrt{1 - k^2} \quad (14-10)$$

Now differentiating with respect to x we have

$$\cos \theta \frac{d\theta}{dx} = \frac{-(1+x)k^2}{\sqrt{1 - (1+x)^2 k^2}} - \sqrt{1 - k^2} \quad (14-11)$$

$$= \frac{-k^2}{\sqrt{\frac{1}{(1+x)^2} - k^2}} - \sqrt{1 - k^2} \quad (14-12)$$

At midband $\theta = 0$ and

$$\frac{d\theta}{dx} = \frac{-k^2}{\sqrt{\frac{1}{(1+x)^2} - k^2}} - \sqrt{1 - k^2} \quad (14-13)$$

For $x \ll 1$, $d\theta/(dx)$ is almost constant at

$$\frac{d\theta}{dx} \approx \frac{-1}{\sqrt{1 - k^2}} = \frac{-k^2 - 1 + k^2}{\sqrt{1 - k^2}} \quad (14-14)$$

$$\frac{d\theta}{dx} \approx \frac{-1}{\sqrt{1 - k^2}} = \frac{-1}{\sqrt{1 - \left(\frac{\lambda_0}{\lambda_c}\right)^2}} \quad (14-15)$$

14.4 MAXIMUM POWER

The maximum power that can be transmitted through a waveguide will depend upon the maximum electric field strength that can exist without breakdown. For the dominant, or $TE_{1,0}$ mode in rectangular waveguide of dimensions a, b ($a > b$), the relation between power and electric field strength is

$$\frac{P}{E^2} = 6.63 \times 10^{-4} a b \left(\frac{\lambda}{\lambda_g} \right) \quad (14-18)$$

If the potential gradient is expressed in volts per centimeter, the dimensions of the guide a and b should be expressed in centimeters for the power P to be given in watts. The maximum field intensity occurs parallel to the narrower dimension of the guide, midway between the side walls, and is independent of the distance from the wide faces of the guide.

Using Equation 1 in Equation 18 and solving for E we find

$$E = \sqrt{\frac{P}{6.63 \times 10^{-4} a b \sqrt{1 - \frac{\lambda^2}{\lambda_c^2}}}}$$

low spurious radiation (grating lobes), both the frequency stability and temperature effect require special consideration to limit their effect on subarray sizing.

In our case

$$\begin{aligned} P &= 6.25 \text{ kilowatts} = 6250 \text{ watts} \\ a &= 0.0995 \text{ meters} = 9.95 \text{ centimeters} \\ b &= 0.0498 \text{ meters} = 4.98 \text{ centimeters} \\ \lambda_c &= 2a = 19.9 \text{ centimeters} \\ \lambda &= 12.25 \text{ centimeters} \end{aligned}$$

Hence

$$E = 491 \text{ volts/centimeter}$$

The peak voltage across the waveguide is

$$V = Eb = 2447 \text{ volts}$$

Section 15
POWER SIZING CONSIDERATIONS

15.1 SOLAR COLLECTOR AREA/BLANKET CONSIDERATIONS

The Power Platform and Power Module solar collectors are based on extrapolations of the SEP solar array under development by NASA. The pertinent characteristics of the SEP array (Reference 1) and the various MDAC concepts are summarized in Table 15-1. The SEP output is 12,500W at the output of the harness at 55 °C (GEO application). The blanket is approximately 125 m^2 (1345 ft^2), although various SEP references show slight variations here ranging from 124 to 126 m^2 (1335.5 to 1356 ft^2). The corresponding specific power is 100 W/m^2 (9.294 W/ft^2). The temperature of the same array operating in LEO is estimated to be $\sim 80^\circ\text{C}$, which reduces the output to 88.77 W/m^2 (8.25 W/ft^2). The Power Module array employs eight SEP wings and produces 89,100W under LEO conditions.

The Power Platform value of 96.95 W/m^2 (9.010 W/ft^2) is based on a higher packing factor for a continuous blanket roll, as contrasted to the small hinged SEP panels. The 456,000-We system (Colume 4) is based on SEP, extrapolated for: (1) temperature, (2) packing factor, and (3) 1.9 Sun blanket illumination. These factors lead to a 152.5 W/m^2 (14.17 W/ft^2) blanket output. However, slightly different numbers were used, as indicated in the table, based on a solar cell efficiency roundoff originally used for this sizing calculations.

Parametric data of solar cell blanket area (and solar collector area for the 2:1 concentrating system) as a function of solar cell efficiency is presented in Figure 15-1. The data is for 456-kWe blanket output to power a $358\text{-kW}_{\text{RF}}$ TA-2 antenna and for a 610-kWe blanket to power a $479\text{-kW}_{\text{RF}}$ version of the antenna. The 2955 m^2 blanket for the 456-kWe case corresponds to Colume 4 of Table 15-1.

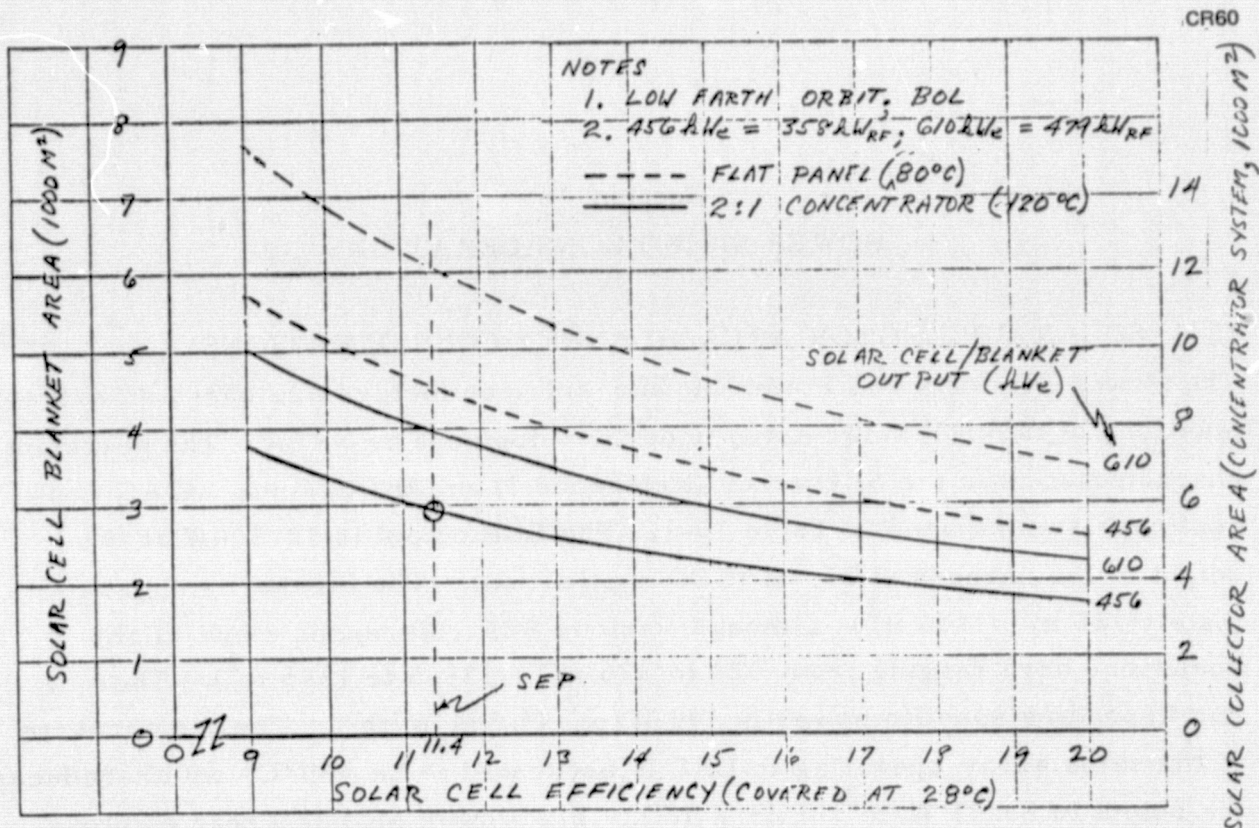


Figure 15-1. Solar Cell Blanket Area Trends

15.2 POWER MODULE SIZING

Parametric electrical power system sizing information is presented for the Power Module in Figure 15-2. The figure shows the solar array output power requirement as a function for the duration of the peak load (TA-1 antenna testing, 75 KWe) with "average load during the peak" (the average of all loads other than the peak load) as a parameter. For example, a 21.5 minute peak of 75 KWe can be supported, along with a 20 KWe average continuous load, by an 89.1 KWe solar array. This operating point is represented by a circle at the intersection of the 20 KWe average load line and the horizontal dotted line at 89.1 KWe. This point represents a reasonable operating point for TA-1 testing for the selected 38 KWe Power Module. The 38 KWe rating is an average power output rating (no peak) as can be seen at the intersection of the 38 KWe average load line with the "Y" axis (peak load duration = 0). In other words, an 89.1 KWe array is required to support this 38 KWe average output system with no peak load.

Table 15-1
SOLAR COLLECTOR PERFORMANCE CHARACTERISTICS⁽¹⁾

	SEP	LEO SEP (e. g., Power Module)	LEO Blanket ⁽³⁾ (e. g., 250-kWe Power Platform)	LEO Blanket (2:1 Concen- tration; e. g., Deployable 456 kWe)
Collector Power Output ⁽²⁾ , We	12, 500	89, 100	250, 000	456, 000
Blanket Area, ft ² /m ²	1354/125	10, 800/1004	27, 750/2, 579	31, 800/2955
Collector Specific Power, W/ft ² :W/m ²	9.294:100	8.25:88.77	9.010:96.95	14.34:154.3
Orbital Regime	GEO	LEO	LEO	LEO
Blanket Tem- perature	55°C	80°C	80°C	120°C
Packing Factor	0.812	0.812	0.887	0.887

(1) Illumination of 1,353 W/m²

(2) BOL; Conditions Specified Herein; 11.4% Cells (AMO at 28°C); Harness Output

(3) 1 Sun Illumination

(4) 2:1 Geometric Concentration; 1.9:1 Actual (1.9 Suns Illumination); Blanket Output

REFERENCE

1. Anon. Solar Array Technology Development for SEP (Solar Electric Propulsion). Mid-Term Report, LMSC D492693, Contract NAS8-31352, 18 January 1977.

The figure also shows the battery output energy requirement (dotted curves, which are read from the right-hand scale). For example, the battery energy requirement for the peak portion of the load is 9.7 KWH per orbit (see circle) for the 21.5 minute peak load described earlier as a representative operating point for TA-1 antenna testing. Note that the circles on both the solar array and the battery curves are on curves labeled 20 KWe average load, a compatible pair.

The far left-hand scale shows power output capability of various numbers of SEP arrays. The 38 KWe Power Module has a total array output capability of 89.1 KWe and employs eight SEP arrays. As noted earlier, this combination can support a 21.5 minute peak and a 20 KWe average load. A six SEP system can support a nine minute peak and 20 KWe of average load. A four SEP system can almost support a 20 KWe average with no peak. Alternatively, a four SEP system can support a nine minute peak and a 10 KWe average load; 10 KWe average is a marginal load capability for Orbiter support requirements.

15.3 POWER PLATFORM SIZING

Parametric data for the Power Platform, similar to the Power Module data presented in Section 15.2, is shown in Figure 15-3. The peak load in this case is the TA-2 antenna at 453 kWe. A 15-minute test once each orbit is a reasonable antenna test requirement. SCB loads can be reduced to approximately 20 kWe average during the intermittent TA-2 test periods. This combination is denoted by the circle at this intersection; it requires a 250 kWe solar array, which is the size selected for the SCB Power Platform. The corresponding battery energy requirement for the peak load is 77.5 kWH.

The 250 kWe solar array has the capability to exceed the energy requirements for a 100 kWe average (power output) system as can be seen by the intersection of the 100 kWe solar array line with the "Y" axis (zero peak load duration); the actual average capability is 106.6 kWe.

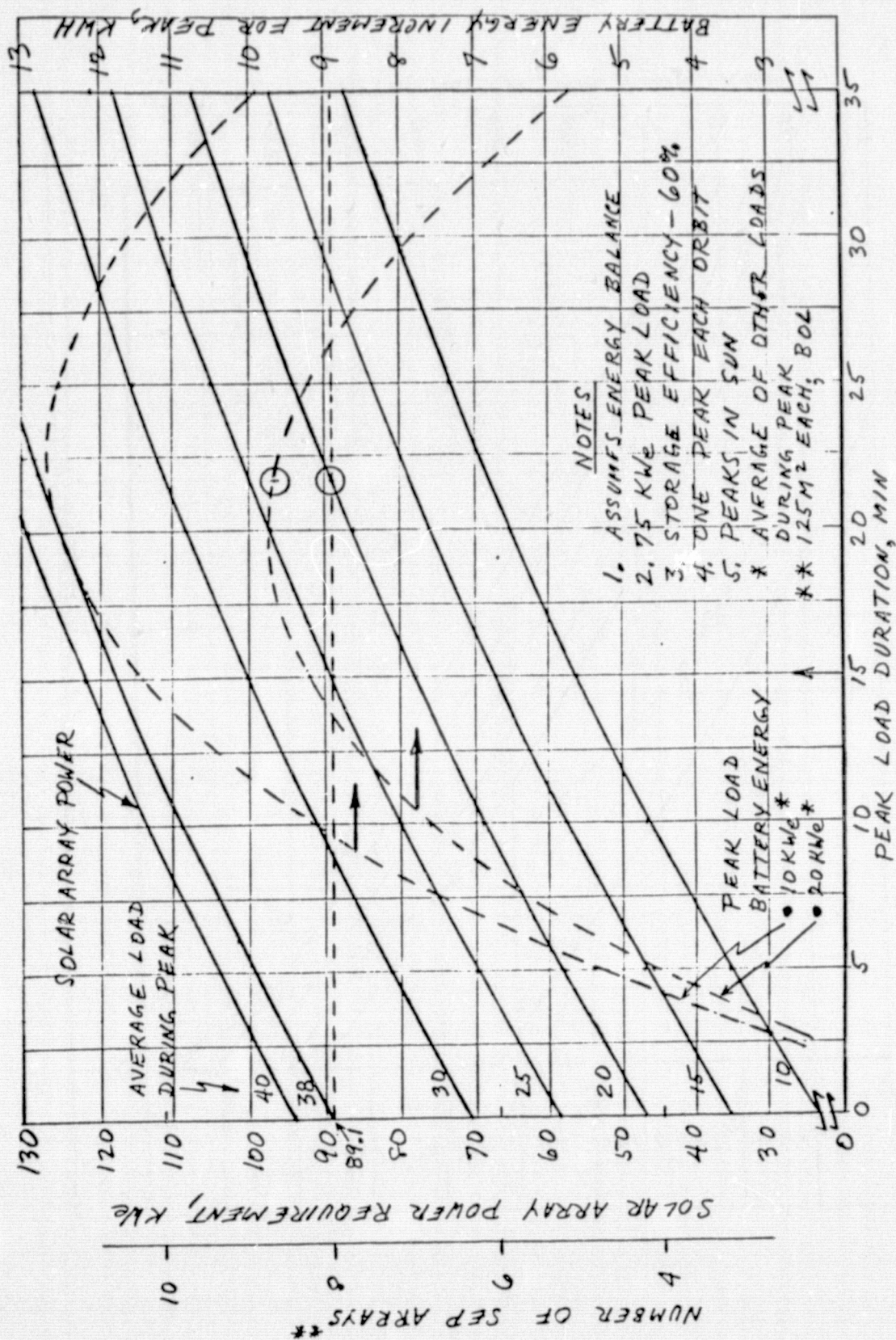


Figure 15-2. 38 kW Power Module Sizing Requirements

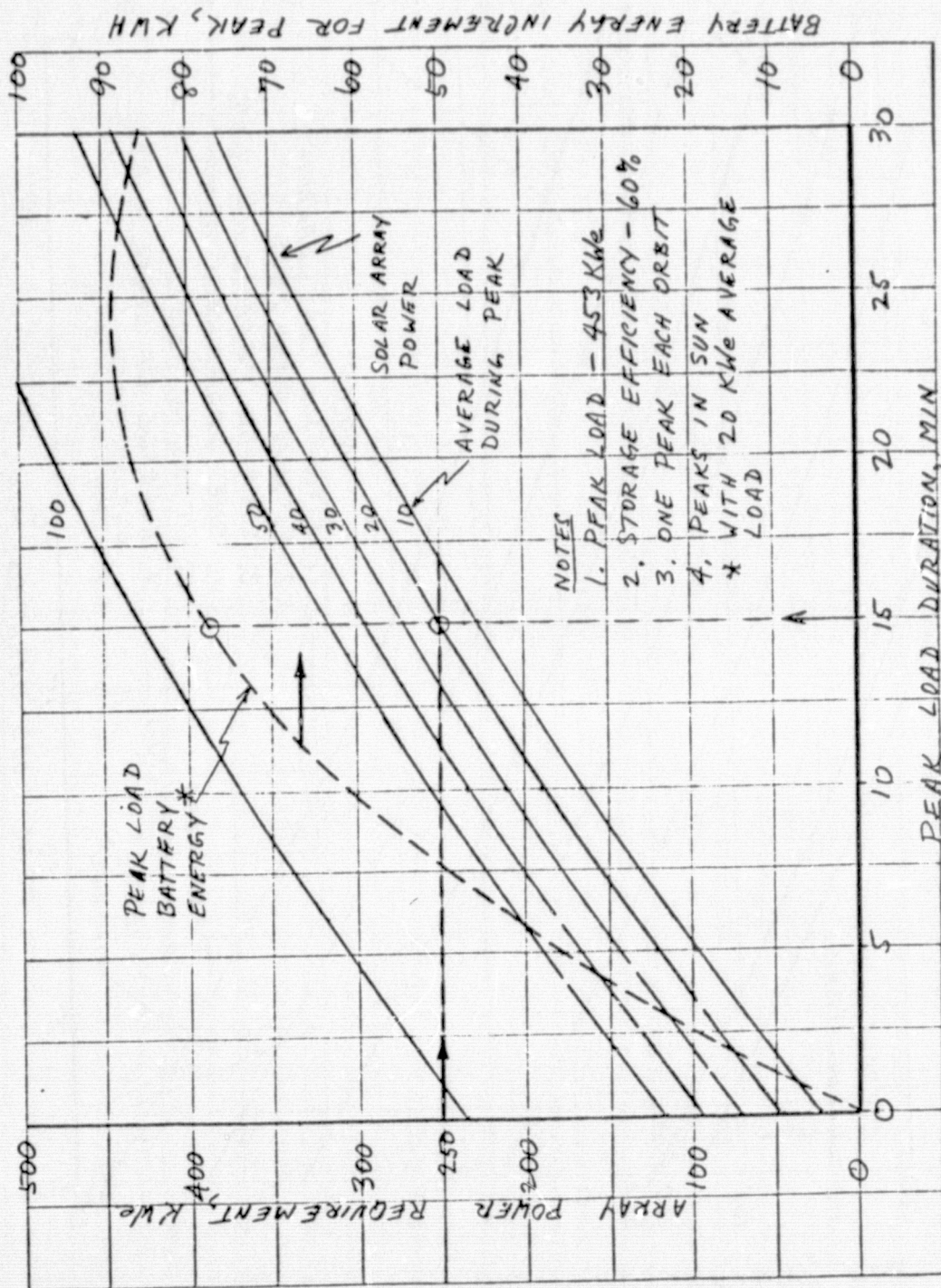


Figure 15-3. Power Platform Solar Array Size Selection (Array Energy Limited)

Section 16 SCB POWER PLATFORM

This section contains drawings generated during Part 3 of this study that are not included elsewhere in the documentation. They include alternative power platform assembly schemes in Subsections 16.1 through 16.6. Subsection 16.7 describes three alternative fabrication/assembly facilities, and Subsection 16.8, fabrication equipment.

16.1 456-kW SINGLE LAUNCH WITH SCM AND CRANE

Figure 16-1 shows the launch packaging arrangement for the assembled 456-kW power platform launched using the SCM. Included are the crane module, EVA airlock, and deployable/retractable strongback. Figure 16-2 depicts the assembly sequence of this configuration. This configuration was replaced by an alternate design later in the study.

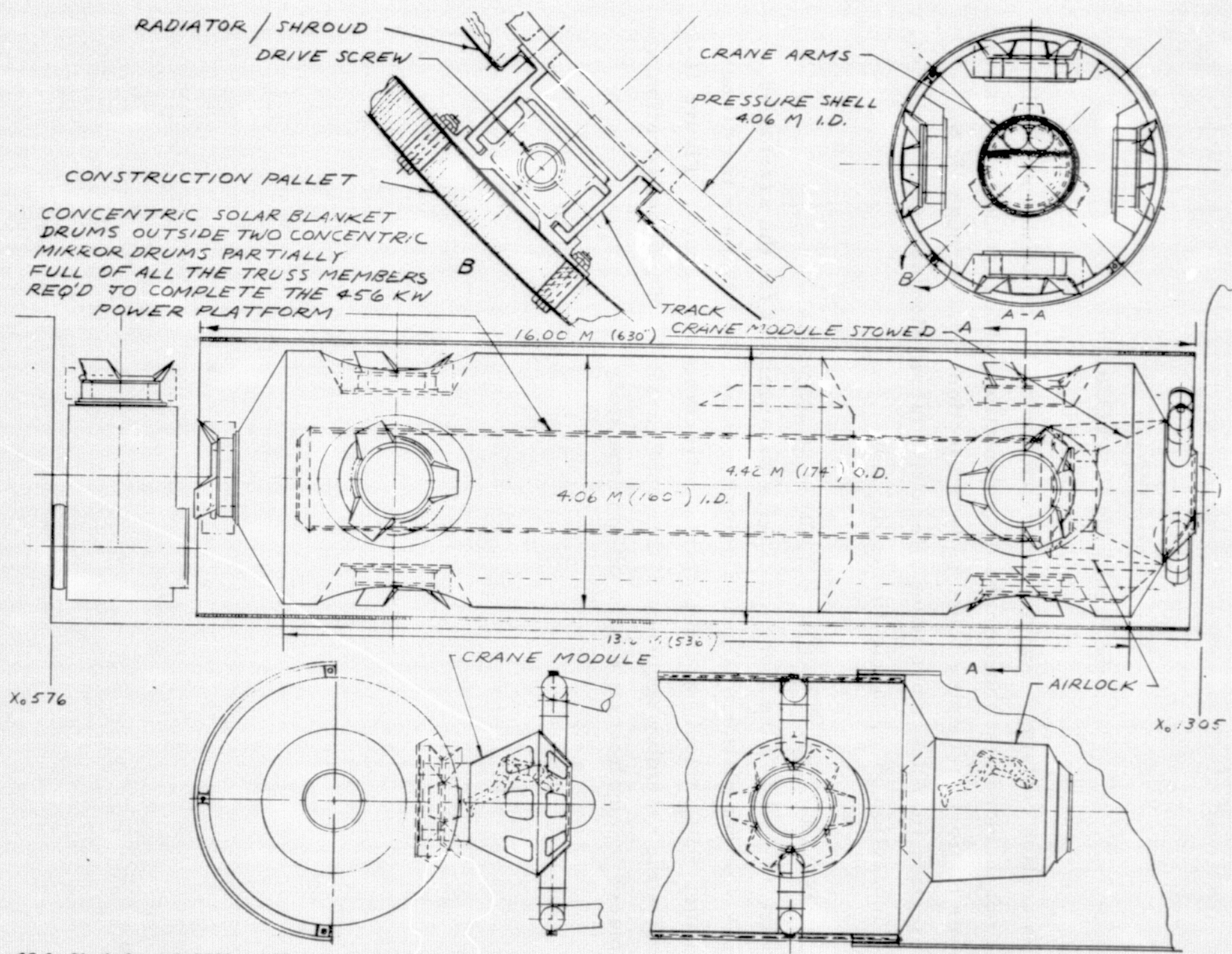
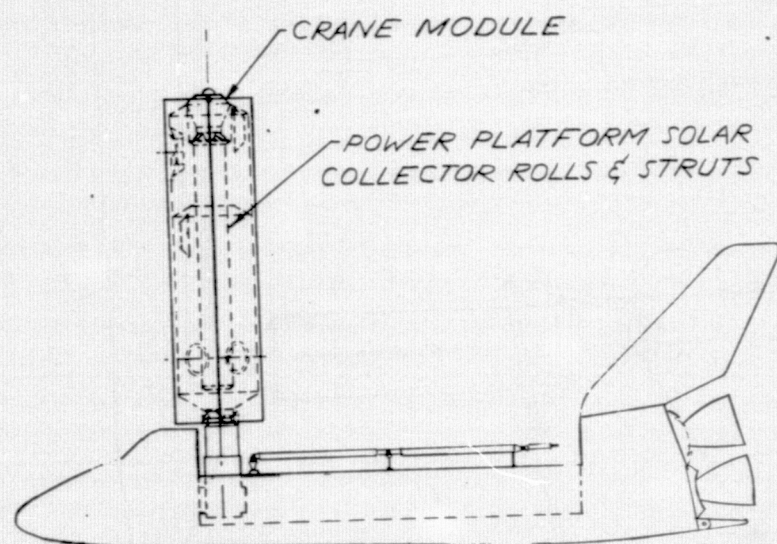


Figure 16-1. Single Launch SCM and PP - Initial Configuration

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- ① ROTATE THE CONSTRUCTION SUPPORT MODULE OUT ON THE PIDAS & BERTH AT THE DOCKING MODULE WITH THE RMS



CONSTRUCTION SUPPORT MODULE WITH
456 KW POWER PLATFORM ASSEMBLED
ON FIRST LAUNCH FROM THE ORBITER

- ② DEPLOY GR/EPOXY PALLET WITH THREE DRIVE SCREWS & HINGE OPEN WITH CRANE MODULE YAW ACTUATORS. SECURE PALLET HINGED JOINT. INSTALL FWD BULKHEAD & COMPARTMENTING BULKHEAD CLOSURES & HATCHES.

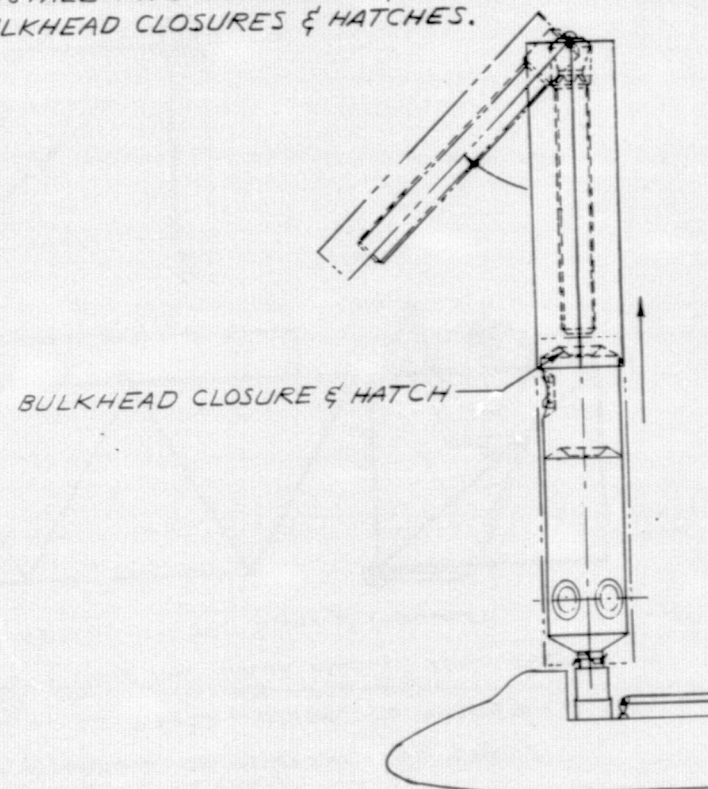
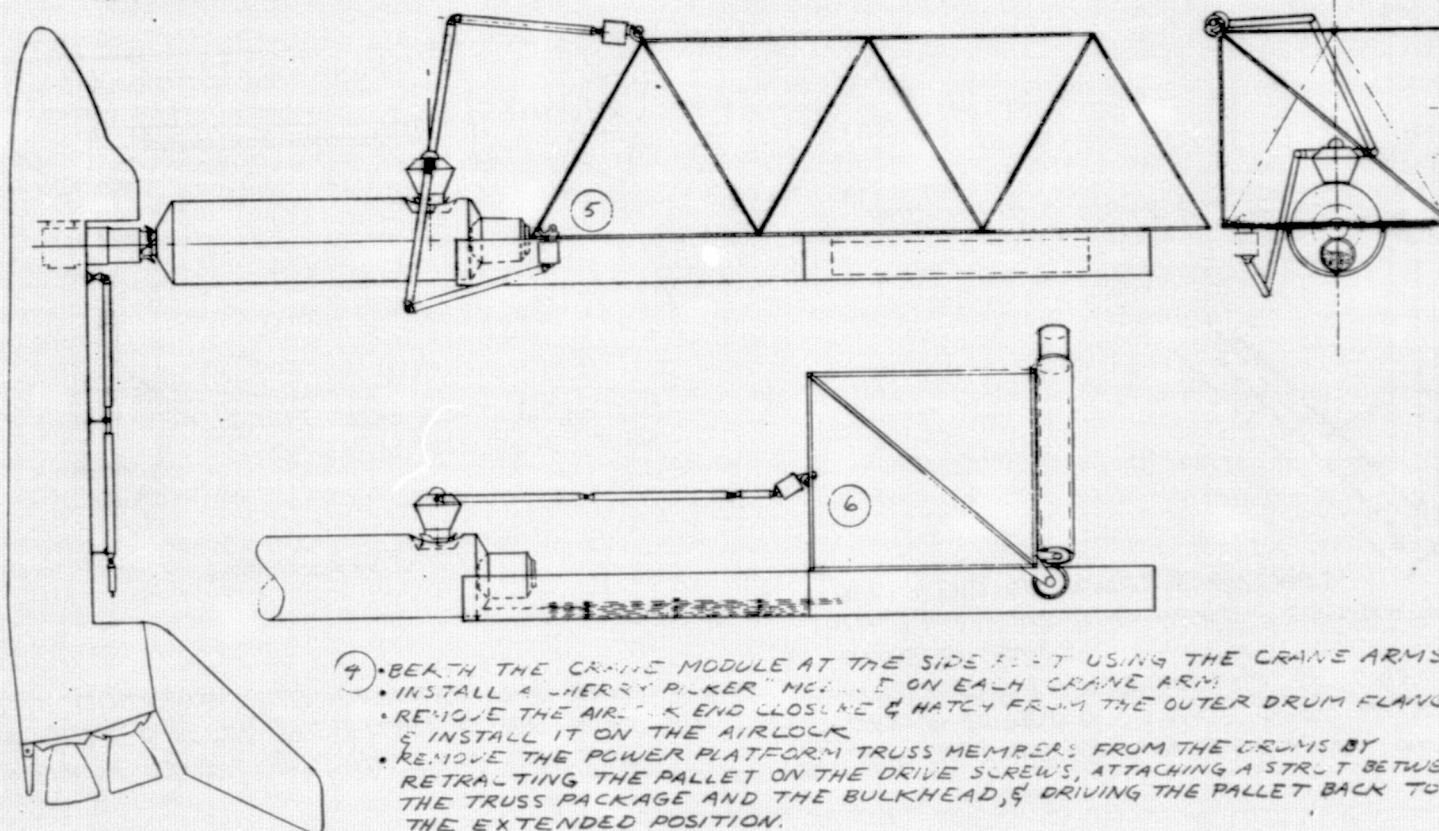


Figure 16-2. Construction Support Module with 456 kW Power Platform Assembled on First Launch from the Orbiter (Sheet 1 of 3)

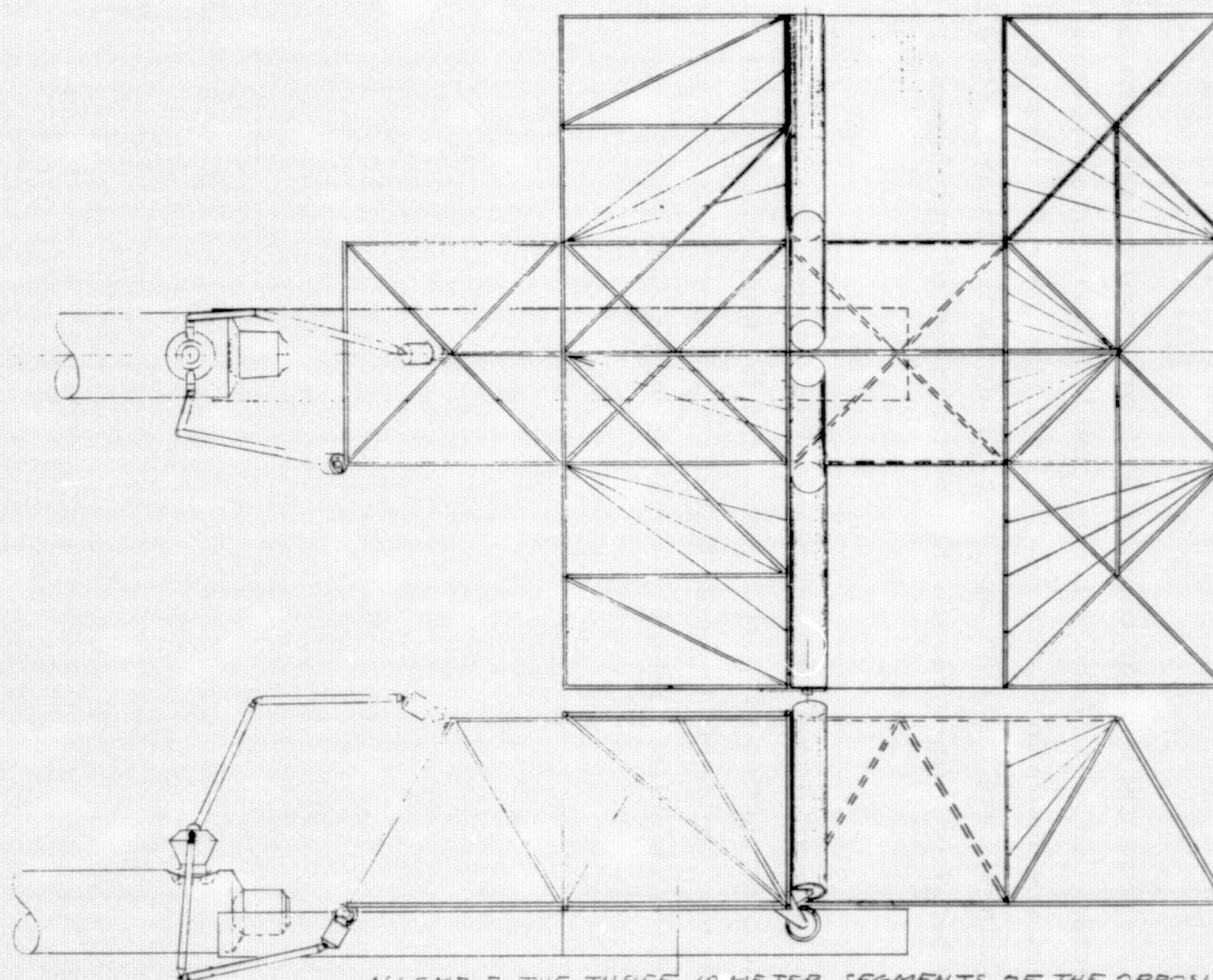
- ③
- REMOVE BOLTS ATTACHING OUTER DRUM FLANGE TO AIRLOCK FLANGE.
 - ROTATE CRANE MODULE BACK TO INITIAL POSITION & REMOVE BOLTS ATTACHING CRANE MODULE TO THE AIRLOCK.
 - MOVE THE AIRLOCK DOWN THE PALLET USING THE GUIDE TRACKS INSTALLED FOR THIS PURPOSE.
 - ROTATE THE AIRLOCK 180° & BOLT IT THE CONSTRUCTION MODULE FORWARD BULKHEAD.
 - RELEASE THE CRANE SPLINED PINS FROM THE PALLET FTGS & MOVE THE CRANE MODULE DOWN THE PALLET USING THE GUIDE TRACKS.
 - RETRIEVE THE RMS CRANE ARMS FROM THE DRUMS & INSTALL ON THE CRANE MODULE.



- ④
- BERTH THE CRANE MODULE AT THE SIDE PLAT USING THE CRANE ARMS.
 - INSTALL A "CHERRY PICKER" MODULE ON EACH CRANE ARM.
 - REMOVE THE AIRLOCK END CLOSURE & MATE FROM THE OUTER DRUM FLANGE & INSTALL IT ON THE AIRLOCK.
 - REMOVE THE POWER PLATFORM TRUSS MEMBERS FROM THE DRUMS BY RETRACTING THE PALLET ON THE DRIVE SCREWS, ATTACHING A STRUT BETWEEN THE TRUSS PACKAGE AND THE BULKHEAD, & DRIVING THE PALLET BACK TO THE EXTENDED POSITION.
- ⑤ ASSEMBLE THE END BEAM/ASSY JIG USING THE "CHERRY PICKERS"
- ⑥ ROTATE THE END BEAM/ASSY JIG 90° & MOUNT THE SOLAR CELL BLANKET & MIRROR DRUMS WITH THE PALLET RETRACTED USING THE "CHERRY PICKERS"

Figure 16-2. Construction Support Module with 456 kW Power Platform Assembled on First Launch
From the Orbiter (Sheet 2 of 3)

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7. ASSEMBLE THE THREE, 10 METER SEGMENTS OF THE OPPOSITE END BEAM, ONE SEGMENT AT A TIME, USING THE CHERRY PICKERS AND ASSY JIG.
8. SHOVE THE COMPLETED SEGMENTS OUT OF THE JIG BY RETRACTING THE PALLET.
9. ROTATE THE END BEAM 90° & ATTACH TO THE MIRRORS AND BLANKETS
10. ASSEMBLE THE CENTER BEAM ONE SEGMENT AT A TIME, WITH THE FIRST SEGMENT ATTACHED TO THE END BEAM, UNTIL THE 19 SEGMENTS ARE COMPLETED.

Figure 16-2. Construction Support Module with 456 kW Power Platform Assembled on First Launch From the Orbiter (Sheet 3 of 3)

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16.2 TA-2C SOLAR POWER MODULE

The TA-2C solar power module (Figure 16-3) is a deployable version of the TA-2. Figure 16-4 shows the complete antenna packaged in the Orbiter payload bay. The 30 solar collector frames are for use later in the deployment of the solar collector. The lower view shows a single row of folded antenna structures with waveguides. The waveguides are already aligned to be parallel and are electrically checked out. Figure 16-5 shows a panel row in both folded and deployed positions. A minimum number of attachments is needed to rigidize the structure in the deployed position.

Figure 16-6 shows one Orbiter bay load of frames and crossbeams for the solar collector. Details of these frames and how their cross-sectional arrangement are shown in Figure 16-7. Note that the reflectors and the solar cell blankets are installed and checked out on the ground before launch. The hinges between frames reduce the orbital installed attachments to a minimum.

Figures 16-8 and 16-9 show the unloading of an Orbiter, the docking of the pallet to the construction support module, and the deployment of the antenna and solar array. It should be noted that three Orbiter loads are required to transport the complete assembly — one for the antenna and two for the solar array.

(Continued on Page 16-14)

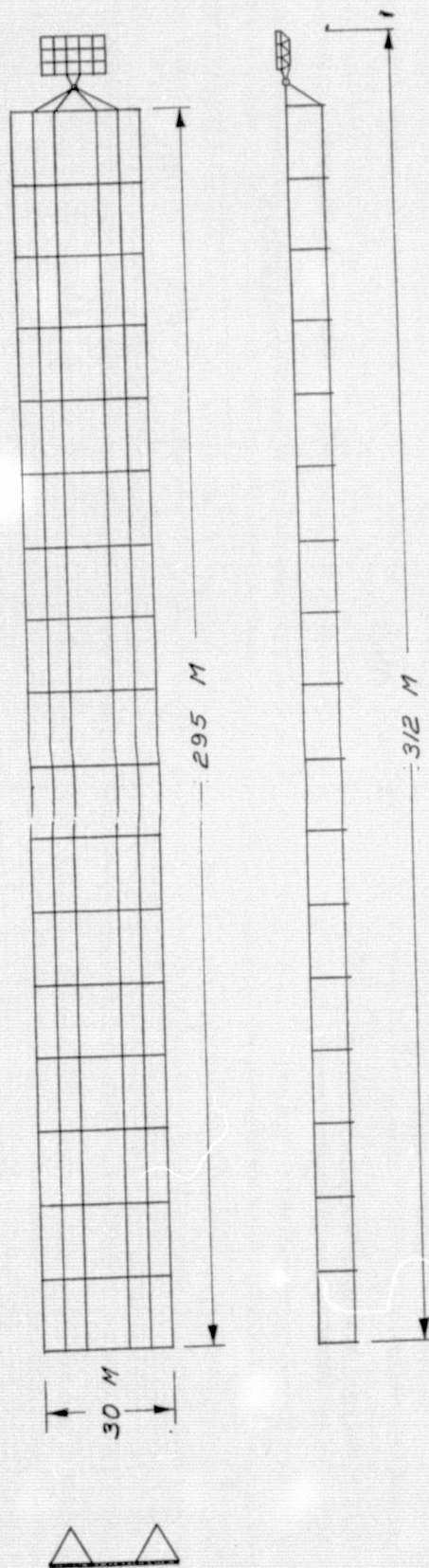


Figure 16-3. TA-2C Solar Power Satellite

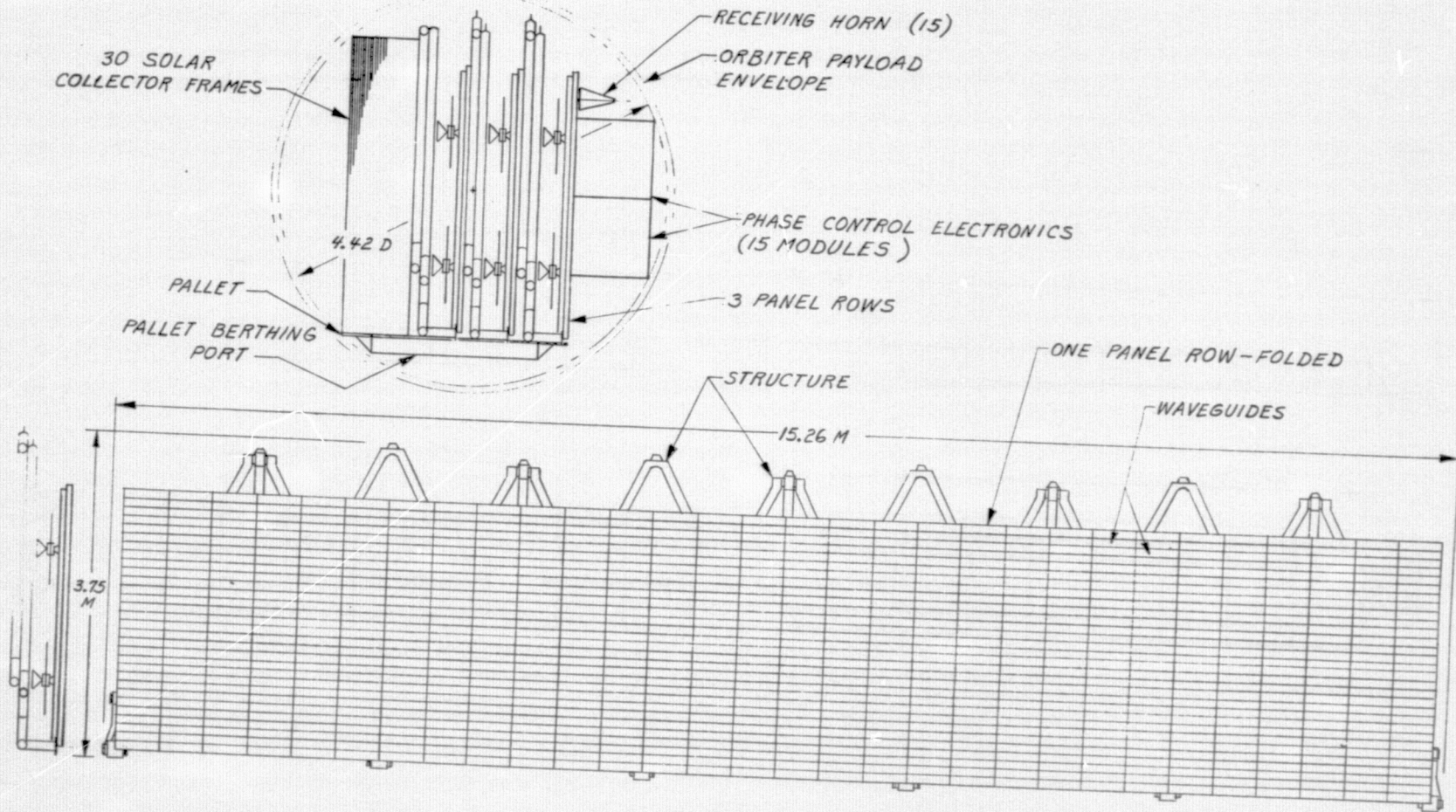


Figure 16-4. TA-2C Antenna - Stowed

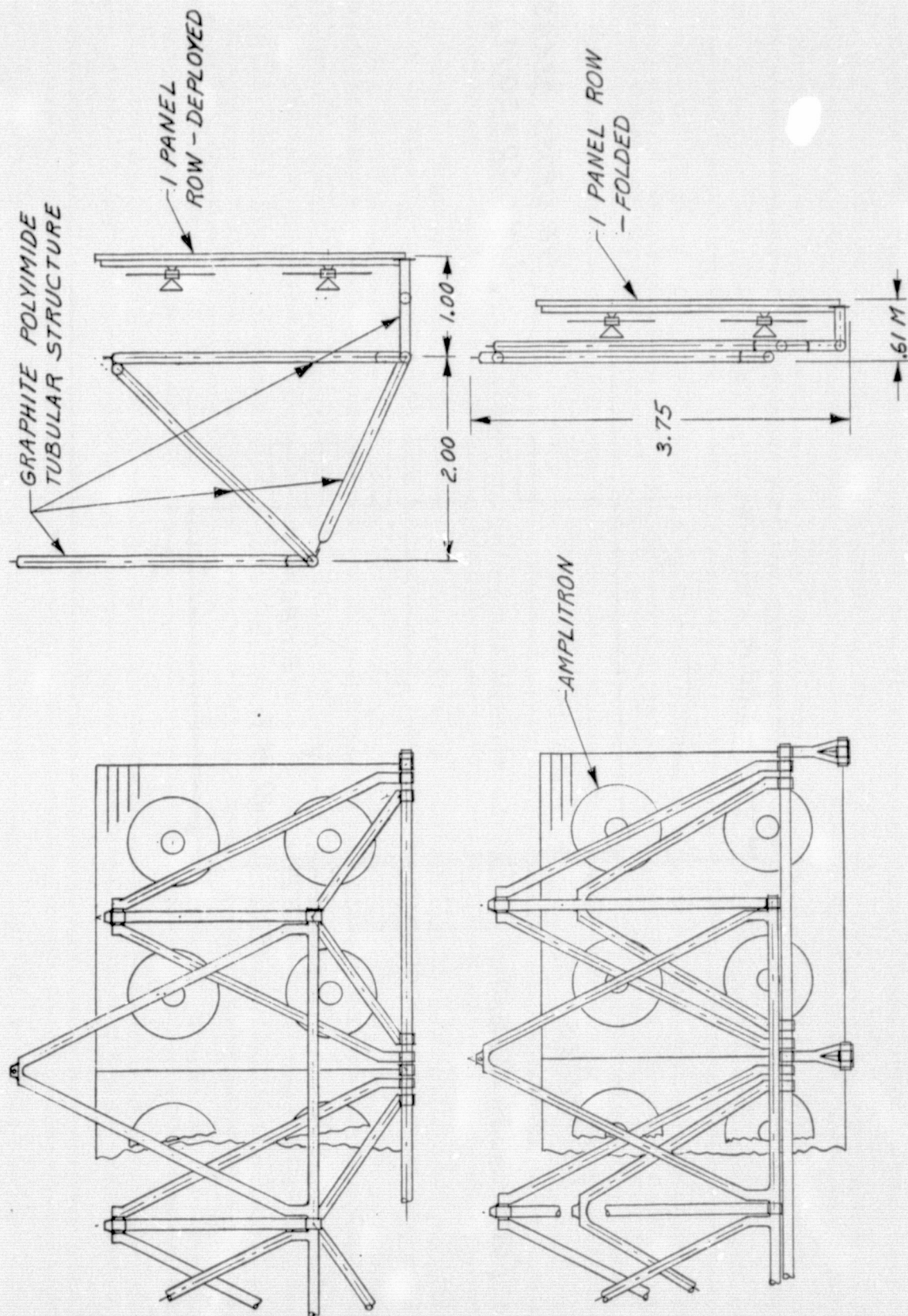


Figure 16-5. TA-2C Antenna - Deployed

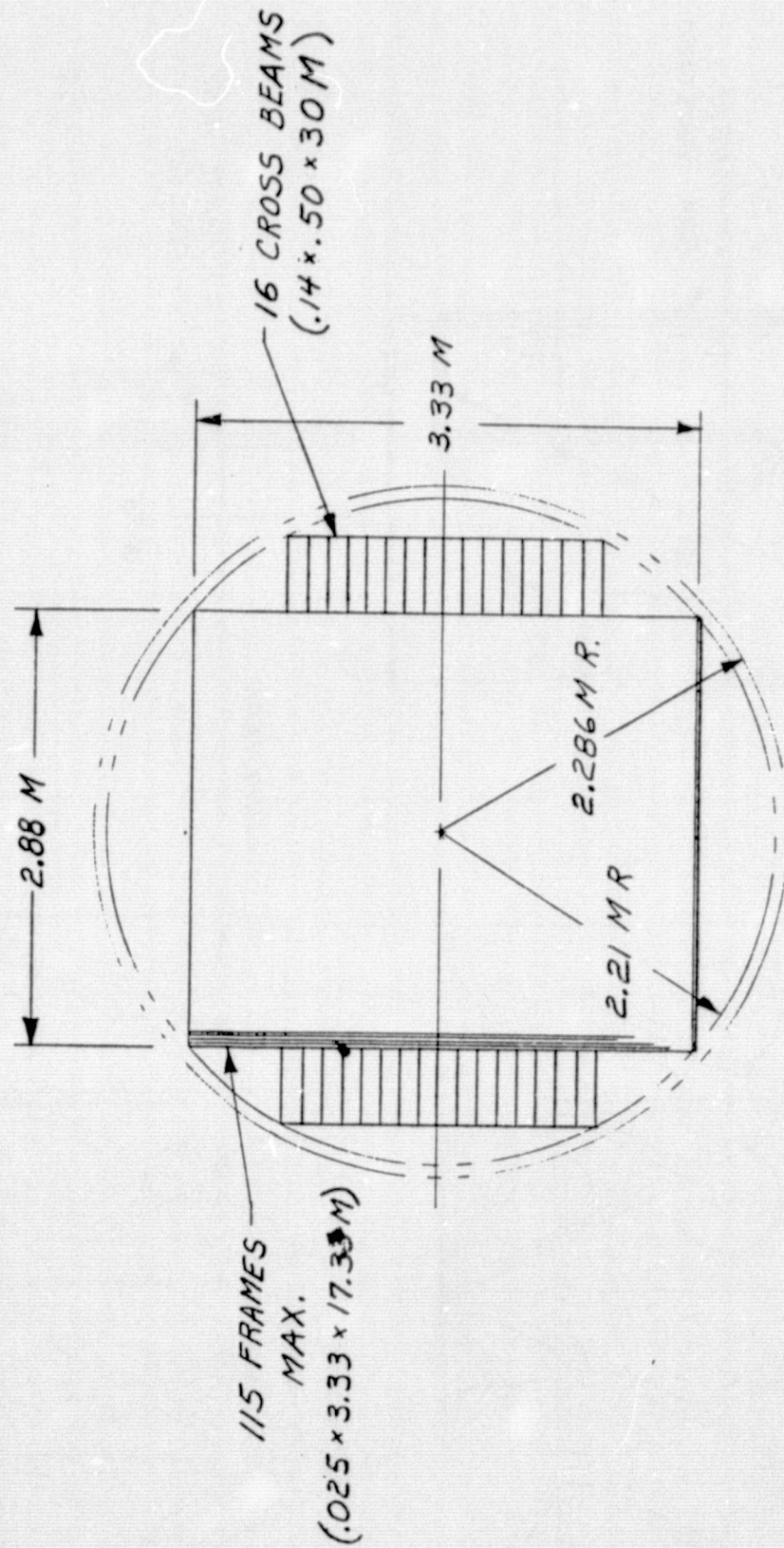


Figure 18-6. TA-2C Solar Collector — Stowed

16-11

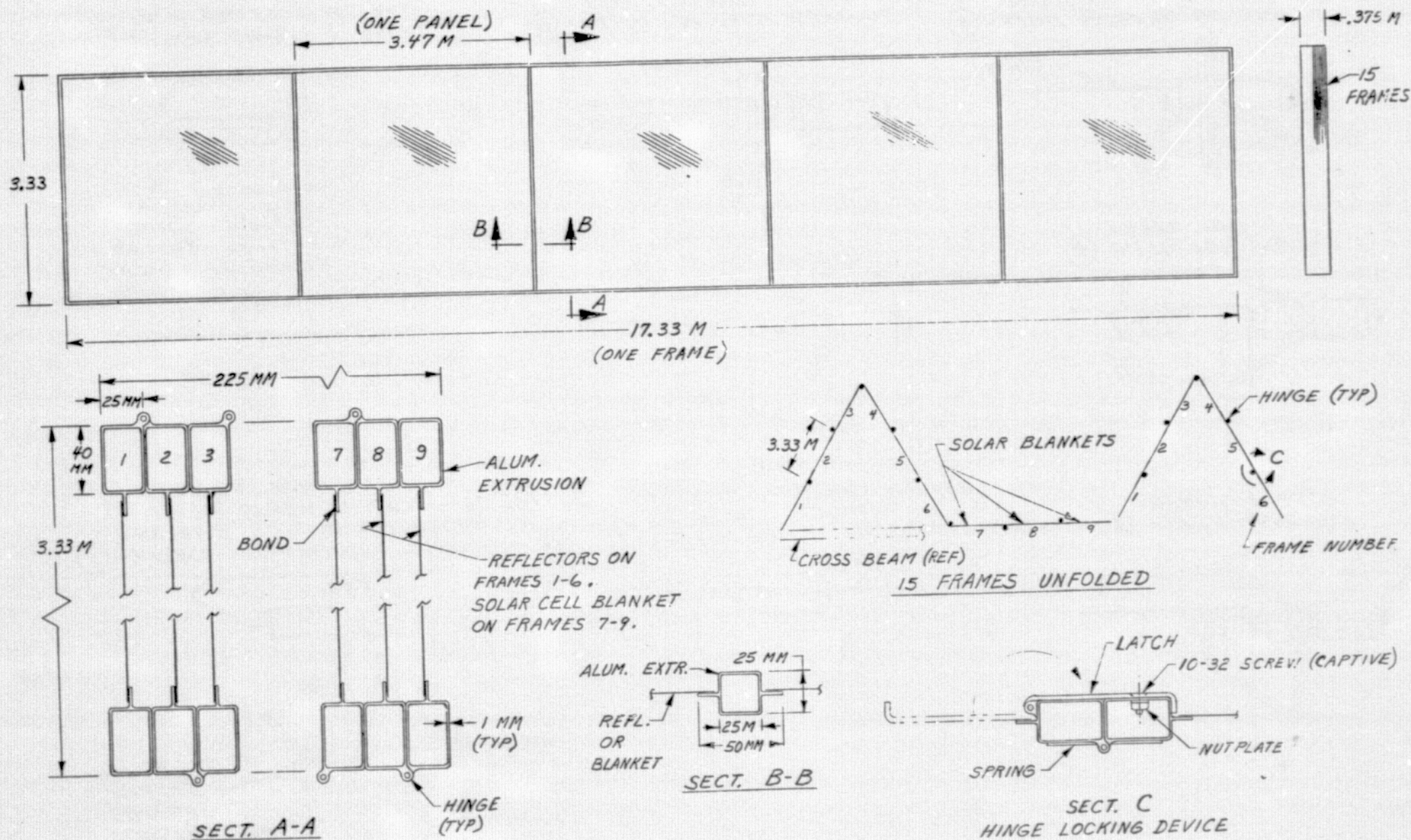


Figure 16-7. TA-2C Solar Collector

CONSTRUCTION
SUPPORT
MODULE

- ① • ATTACH END EFFECTOR TO PAYLOAD BAY
- RETRACT DOCKING ACTUATORS & LOCK
- ATTACH OTHER EFFECTOR TO PALLET
- RELEASE PIDAS

- ② • BERTH PALLET TO CONSTRUCTION
SUPPORT MODULE
- ATTACH CHERRY PICKER
TO CRANE

- ③ DEPLOY ASSEMBLY
BEAMS & LOCK

- ④ • REMOVE FIRST PANEL ROW &
ATTACH TO BEAMS
- UNFOLD PANEL ROW &
JOIN STRUCTURE

- ⑤ DITTO FOR SECOND
PANEL ROW

- ⑥ DITTO FOR THIRD
PANEL ROW

- ⑦ REMOVE GIMBAL & MOUNT
FROM PALLET & INSTALL

- ⑧ • DITTO FOR PHASE CONTROL
ELECTRONICS
- PERFORM CHECKOUT

- ⑨ • STOW ANTENNA ON SIDE OF
MODULE
- STOW 30 COLLECTOR FRAMES
ON BEAMS

FRAMES

Figure 16-8. TA-2C Antenna Deployment Sequence

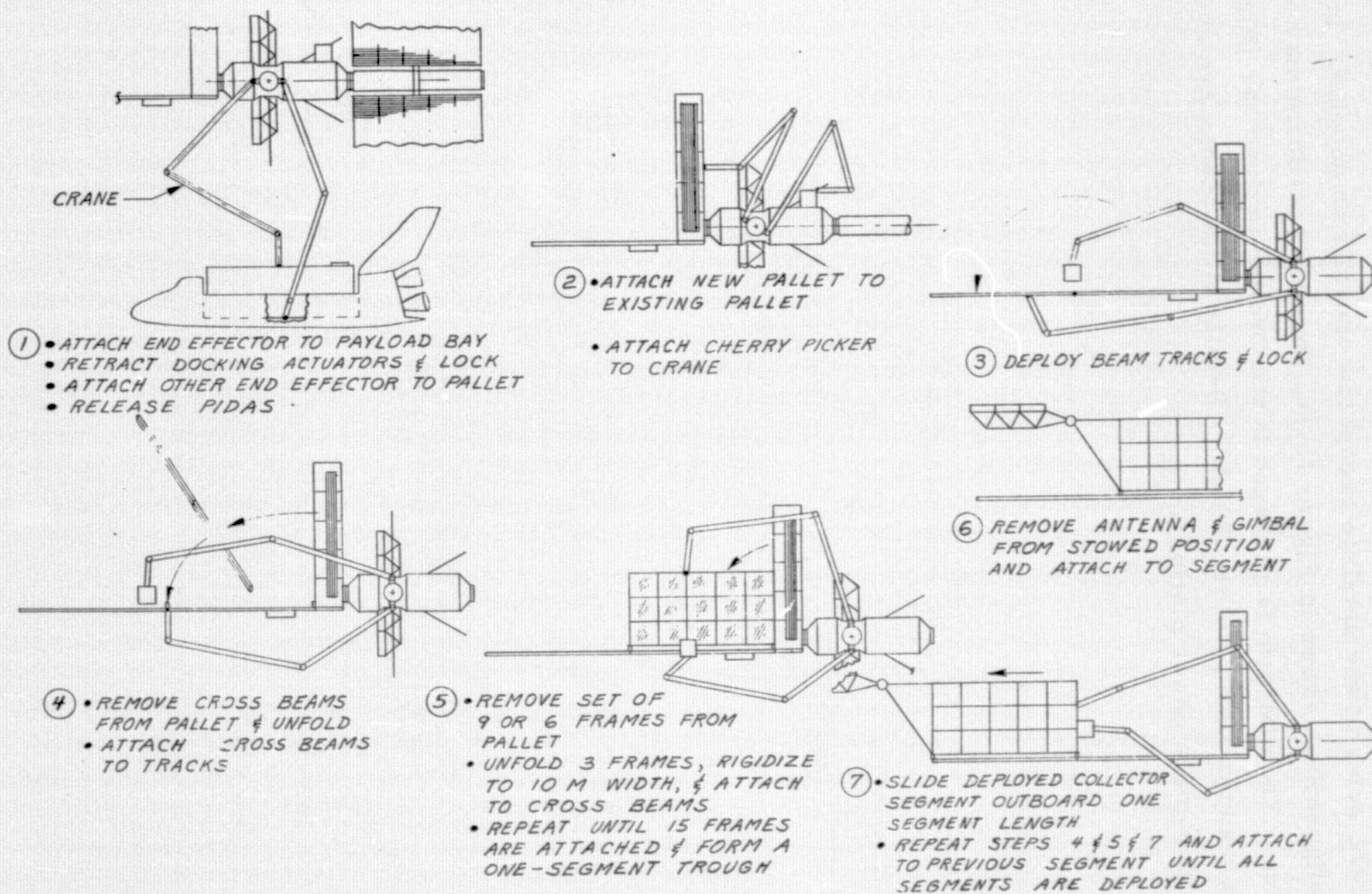


Figure 16-9. TA-2C Solar Collector Deployment Sequence

(Continued from Page 16-6)

16.3 DEPLOYABLE 456-kW POWER PLATFORM

The deployable 456-kW power platform is similar in construction to the TA-2C solar power satellite except that there is no antenna. In the trail configuration, shown in Figure 16-10, the power platform may be in one piece. The platform is packaged in the Orbiter payload bay (Figure 16-11) in a manner similar to the TA-2C. Details of the tracks are shown later in the two-piece power platform.

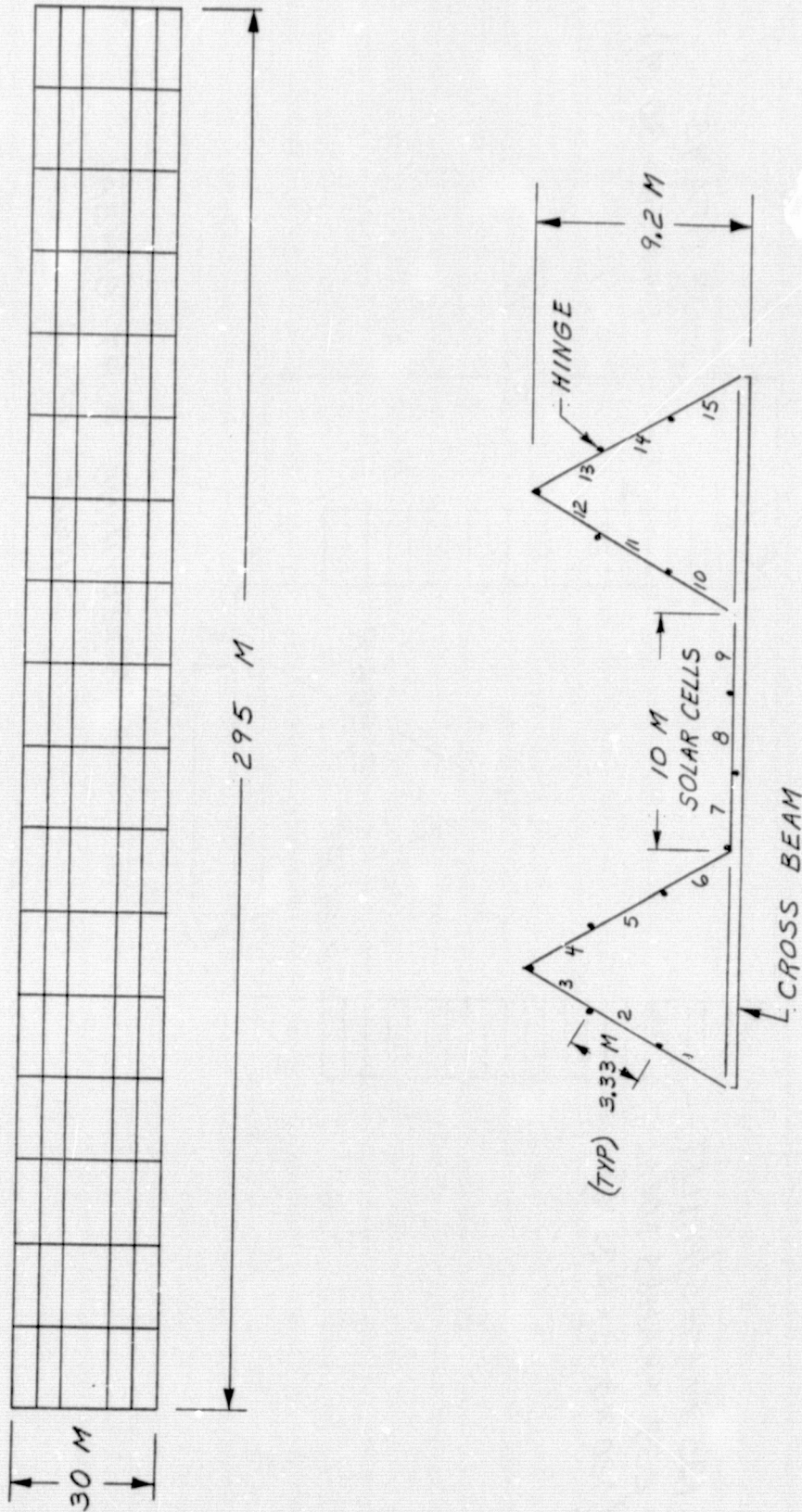
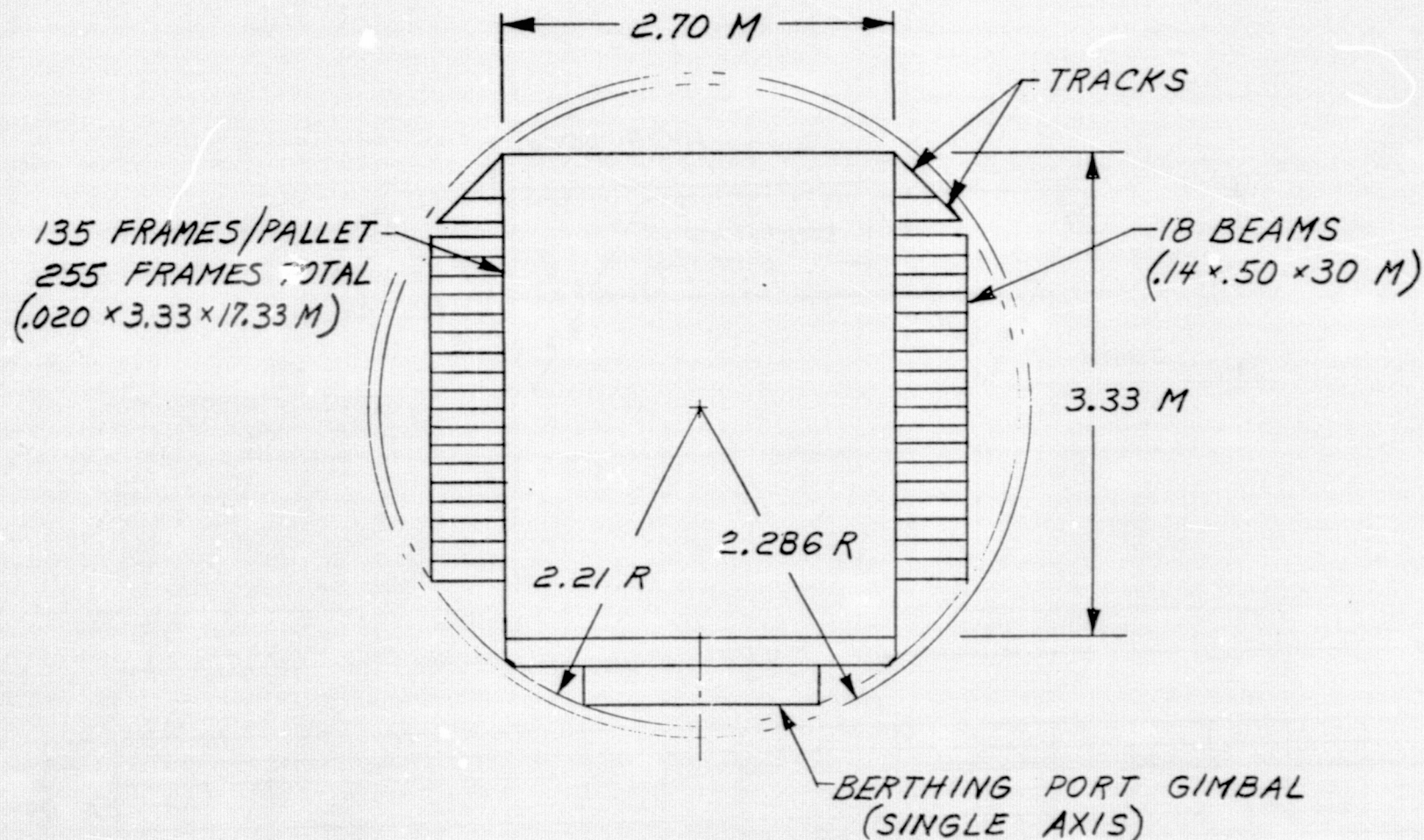


Figure 16-10. Deployable 456 kW Power Platform — Trail Configuration



16-16

Figure 16-11. Deployable 456 kW Power Platform — Trail Configuration/Stowed

16.4 150-kW ONE PIECE POWER PLATFORM

When it was determined that 150 kW would be the optimal size for some configurations, the small one-piece power platform evolved (Figure 16-12). The construction and deployment are similar to those for the TA-2C and the 456-kW power platform. Since the size is smaller, the entire platform can be transported in one Shuttle trip (Figure 16-13).

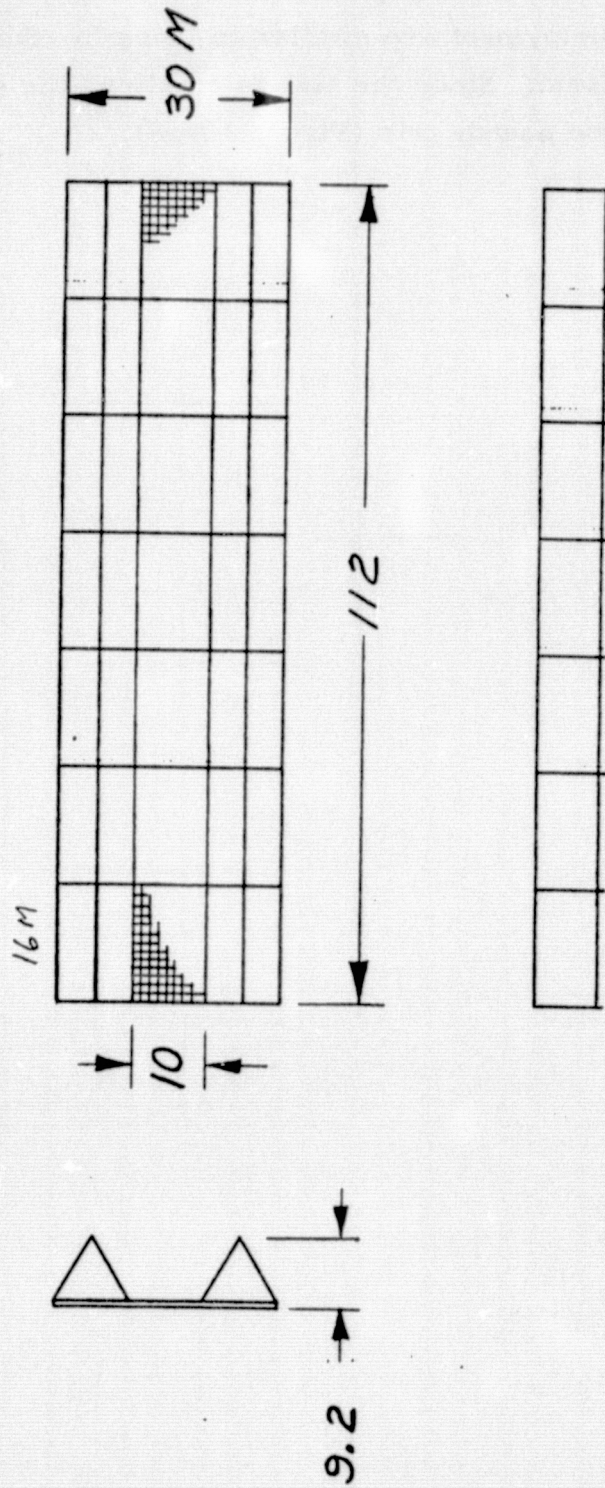


Figure 16-12. One-Piece 150 kW Power Platform - Deployed

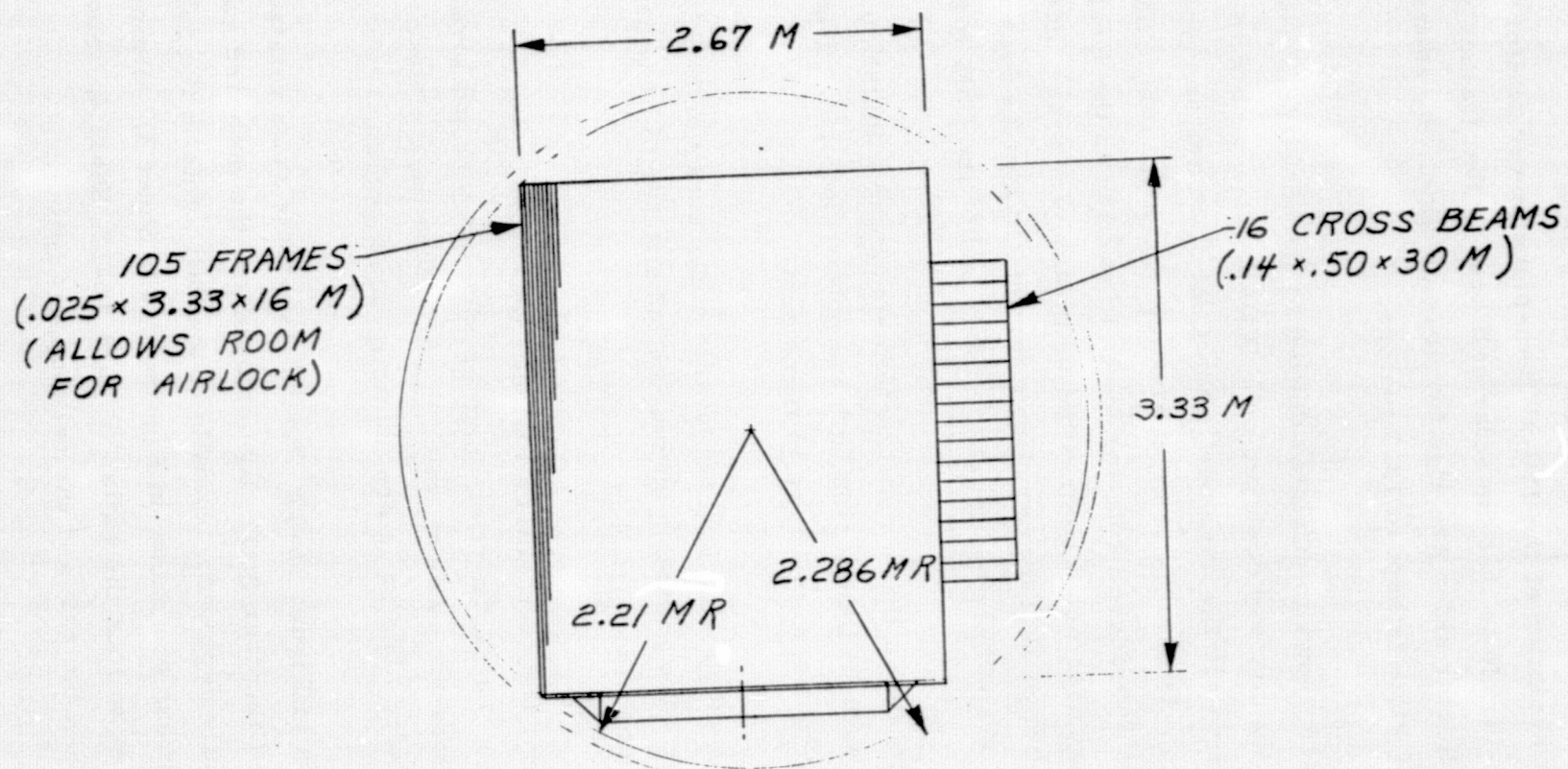


Figure 16-13. One-Piece 150 kW Power Platform — Stowed

16.5 150-kW TWO-PIECE POWER PLATFORM

The latest evolution of the 150-kW two-piece power platform is shown in Figure 16-14. Note that the solar cell area is two frames wide (6.7 meters) and the reflector sides are also two frames wide. By squaring off the sides, the overall width is kept to 13.3 meters.

Figure 16-15 shows the Orbiter docked to the construction support module (CSM) and the payload deployed on a PIDAS. One part of the gimbal joint — that portion within the envelope of the CSM diameter and which pivots 180 degrees — is carried to orbit already attached to the CSM. The other portion, the 360-degree rotating joint, is carried in the first payload and is the first part assembled to the CSM by means of a crane (not shown). This figure also shows that there is room for the Orbiter to dock and for the attachment of another module at the other end of the CSM if necessary.

Figure 16-16 shows that a construction shack and a logistics module may be added later without interfering with the power platform. The cross-section of the power platform is shown in more detail.

Figure 16-17 shows a typical deployable solar array on a pallet. A berthing port is shown but a gimbal joint may be substituted to be consistent with the configuration of Figure 16-16.

Details of the rails (sometimes called tracks) are shown in Figure 16-17, Section C-C. The rectangular tubes provide bending and torsional stiffness. The tee sections are rails to which the crossbeams are attached and along which they slide. The rails are shown folded together.

The two views in Figure 16-17, Sheet 2, show the rails rotated 90 degrees but still folded together (partially deployed and the outboard rails folded out and locked (fully deployed). Details of the rail joints in the folded and deployed positions are shown in Figure 16-17, Sheet 3. Minimal astronaut effort is needed to lock the rails. Figure 16-17, Sheet 4, shows details of the joint between the rail and the rail support fitting.

(Continued on Page 16-28)

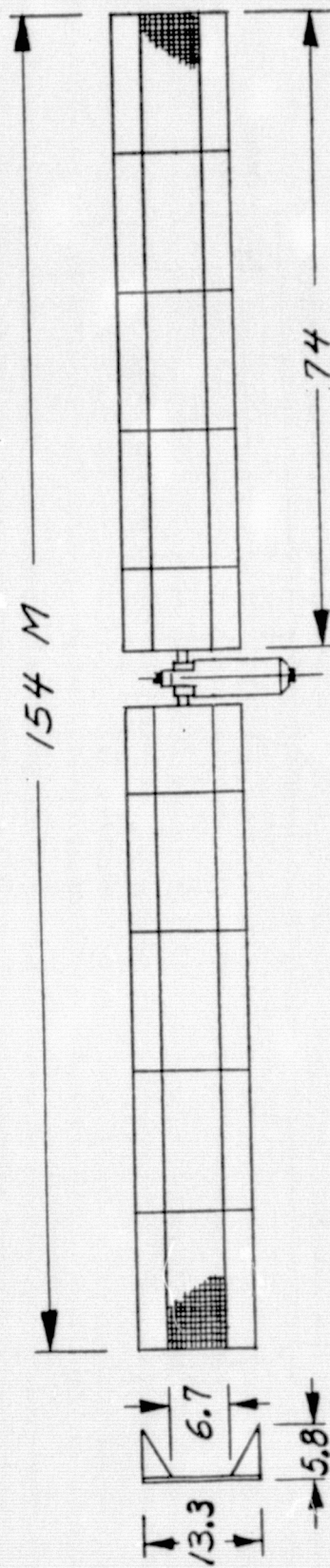


Figure 16-14. Two-Piece 150 kW Power Platform — Plan View

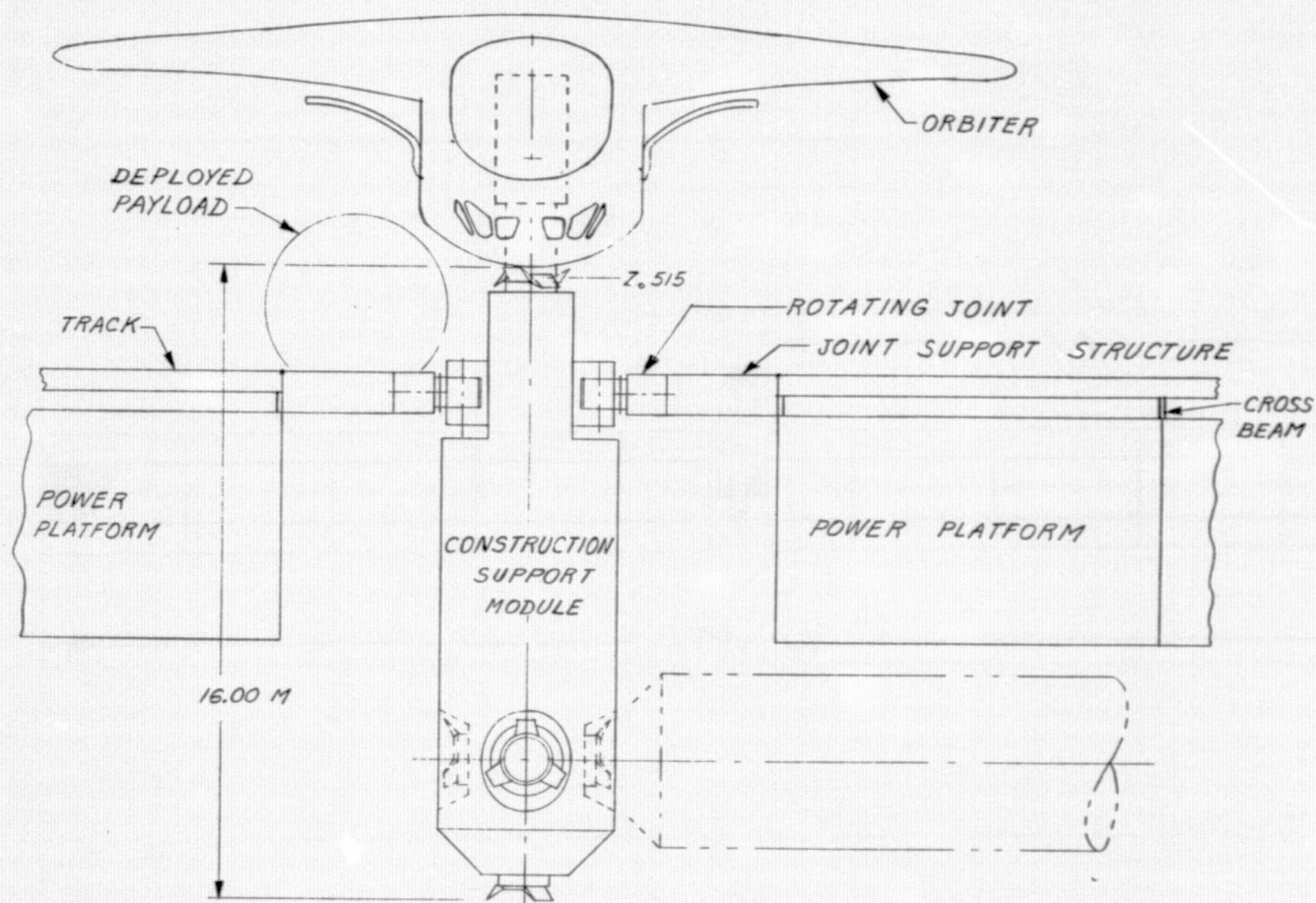


Figure 16-15. Two-Piece 150 kW Power Platform - Option 1

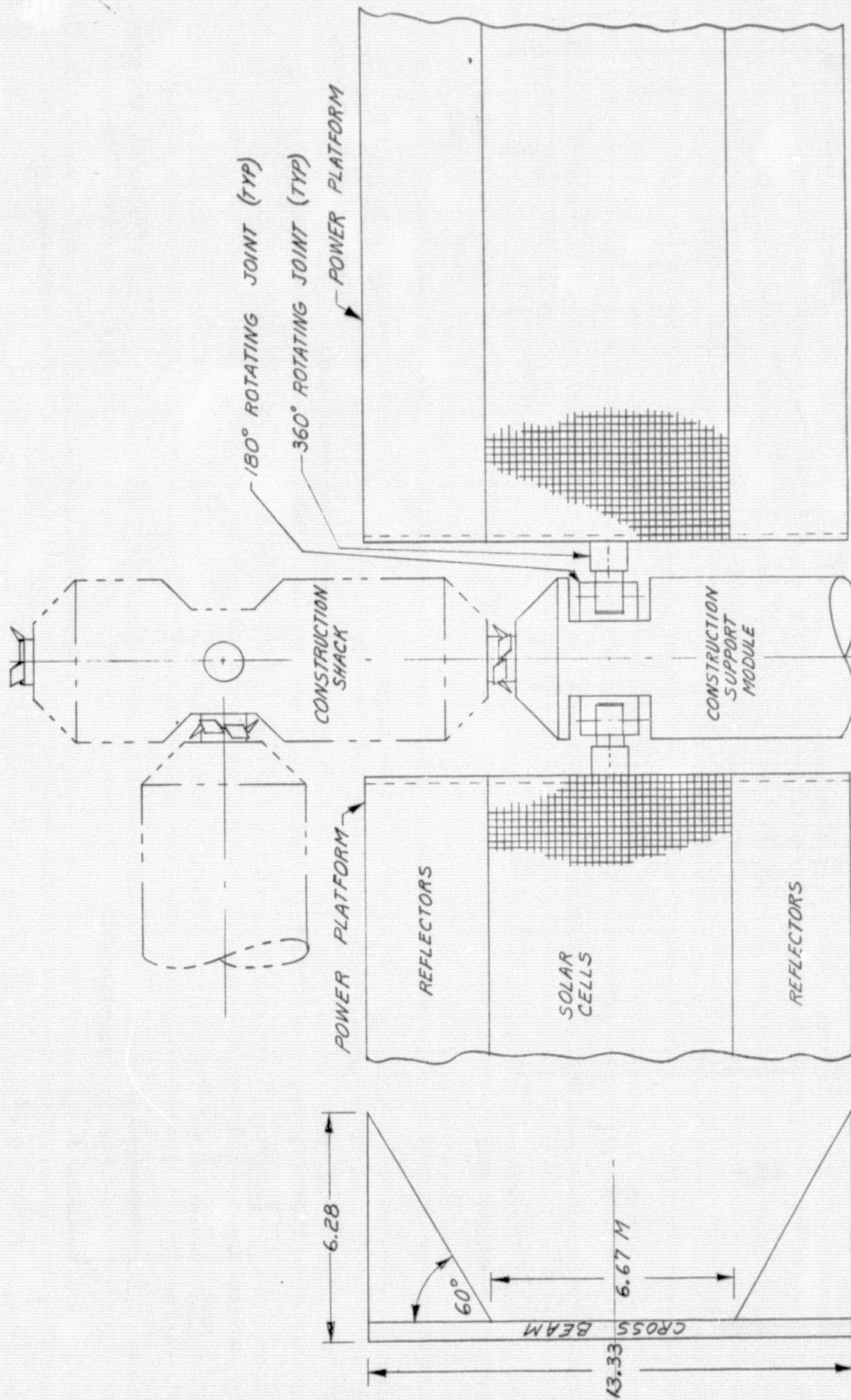


Figure 16-16. Two-Piece 150 kW Power Platform — Option 2

CR60

16-24

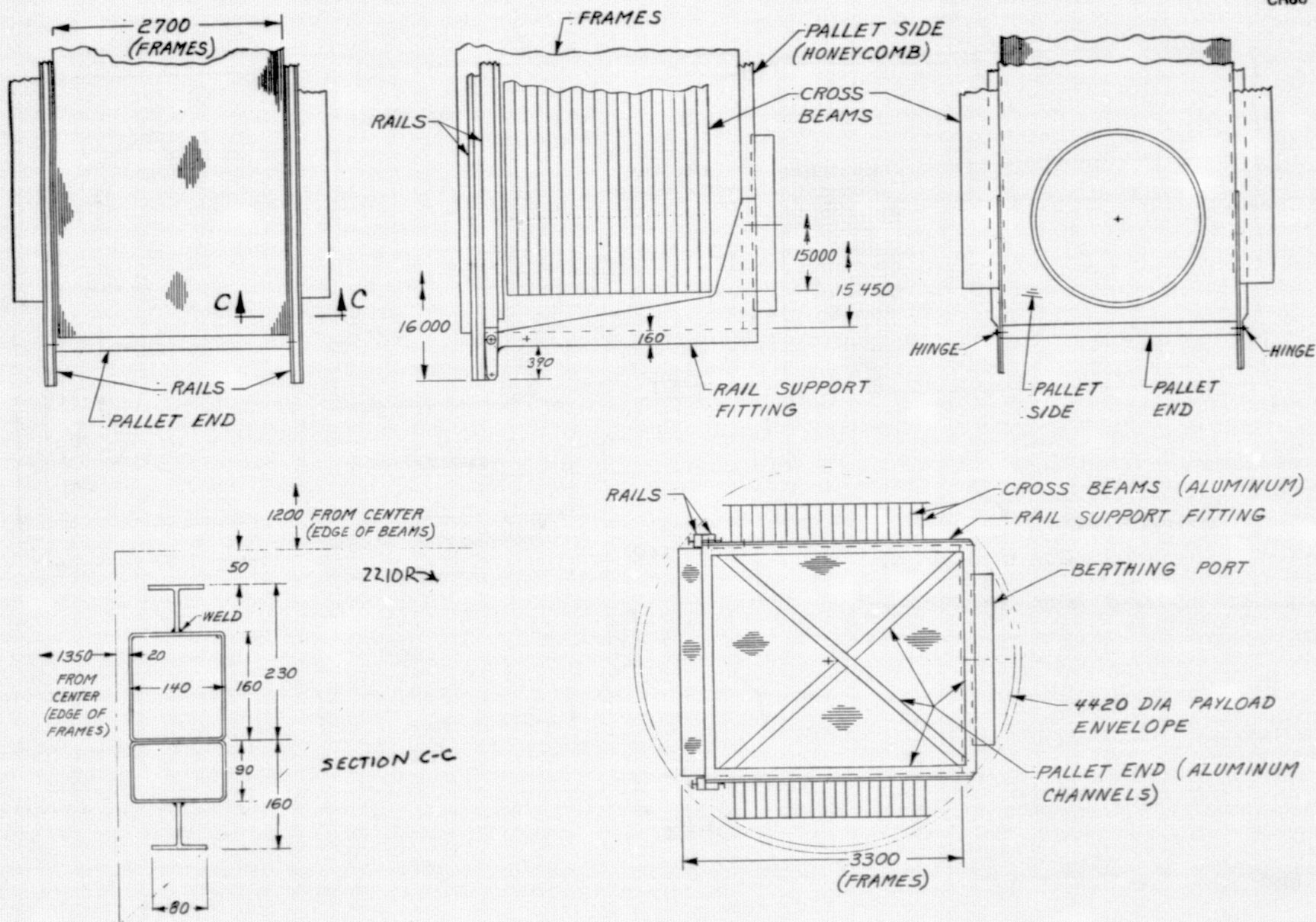
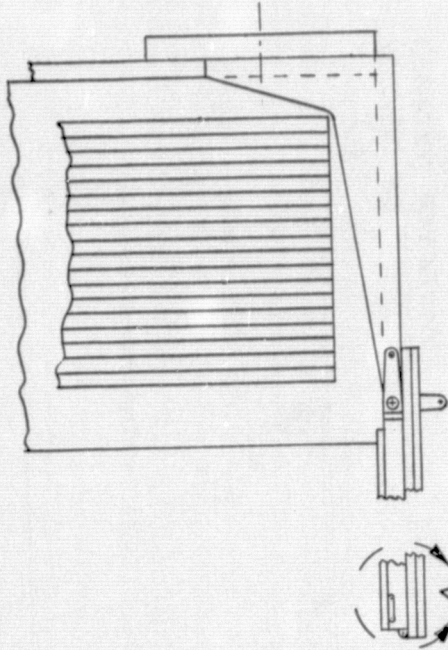
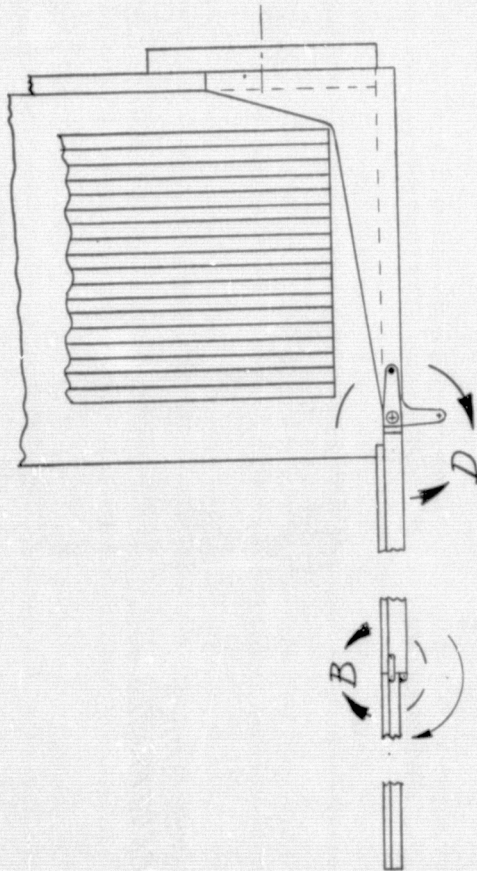
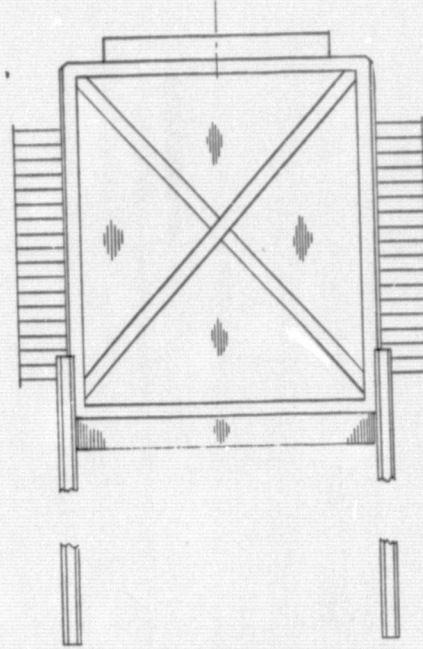


Figure 16-17. Deployable Solar Array on Pallet (Sheet 1 of 4)

CR60



RAILS PARTIALLY DEPLOYED



RAILS FULLY DEPLOYED

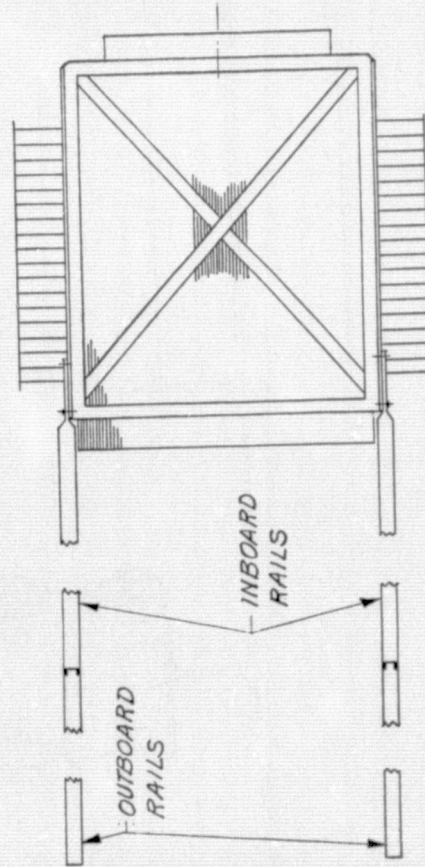


Figure 16-17. Deployable Solar Array on Pallet (Sheet 2 of 4)

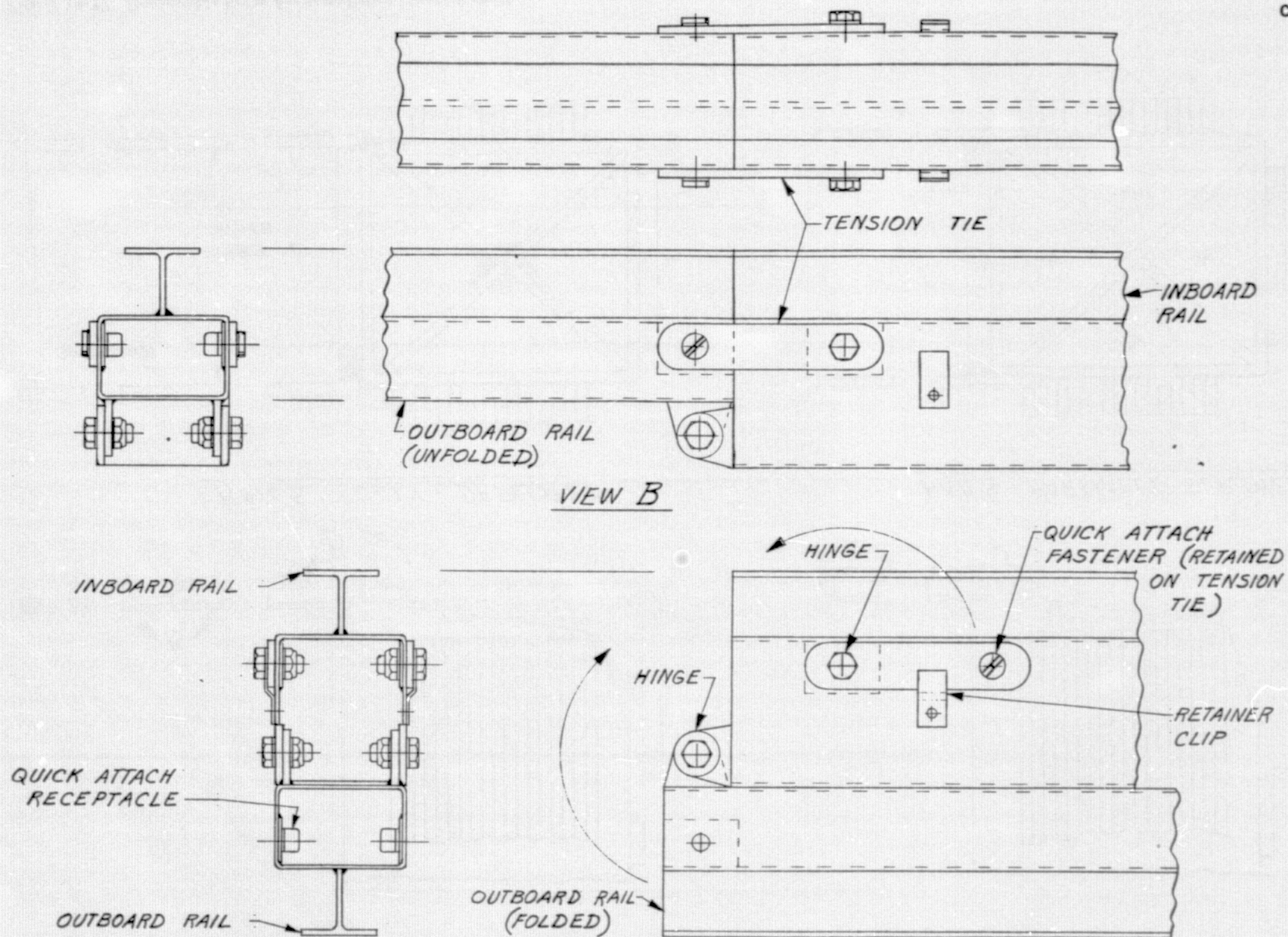


Figure 16-17. Deployable Solar Array or Pallet (Sheet 3 of 4)

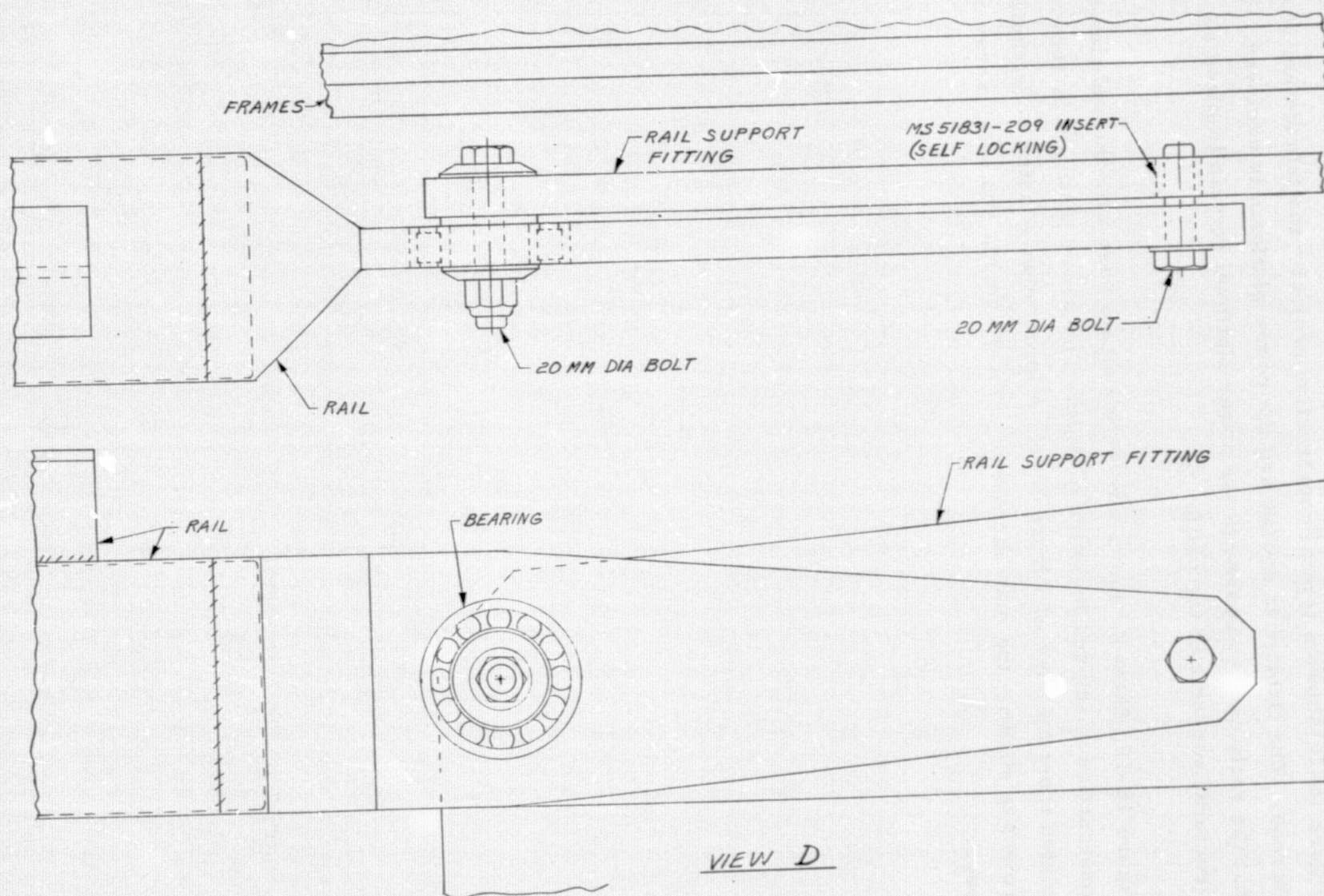
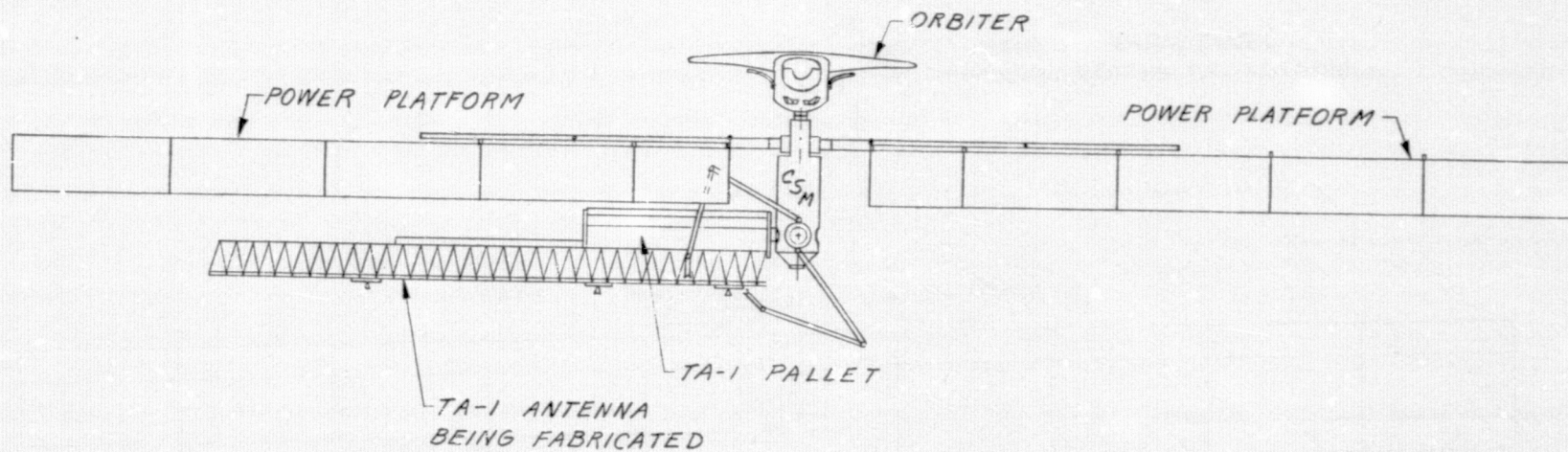


Figure 16-17. Deployable Solar Array on Pallet (Sheet 4 of 4)

(Continued from Page 16-20)

16.6 150kW POWER PLATFORM DOUBLE GIMBAL, SHUTTLE-TENDED
Figure 16-18 shows a typical construction project - the fabrication of a TA-1 antenna. The Orbiter is docked and the power platform is deployed. Note that the rails used to assemble the power platform remain in place as structural supports of the platform.

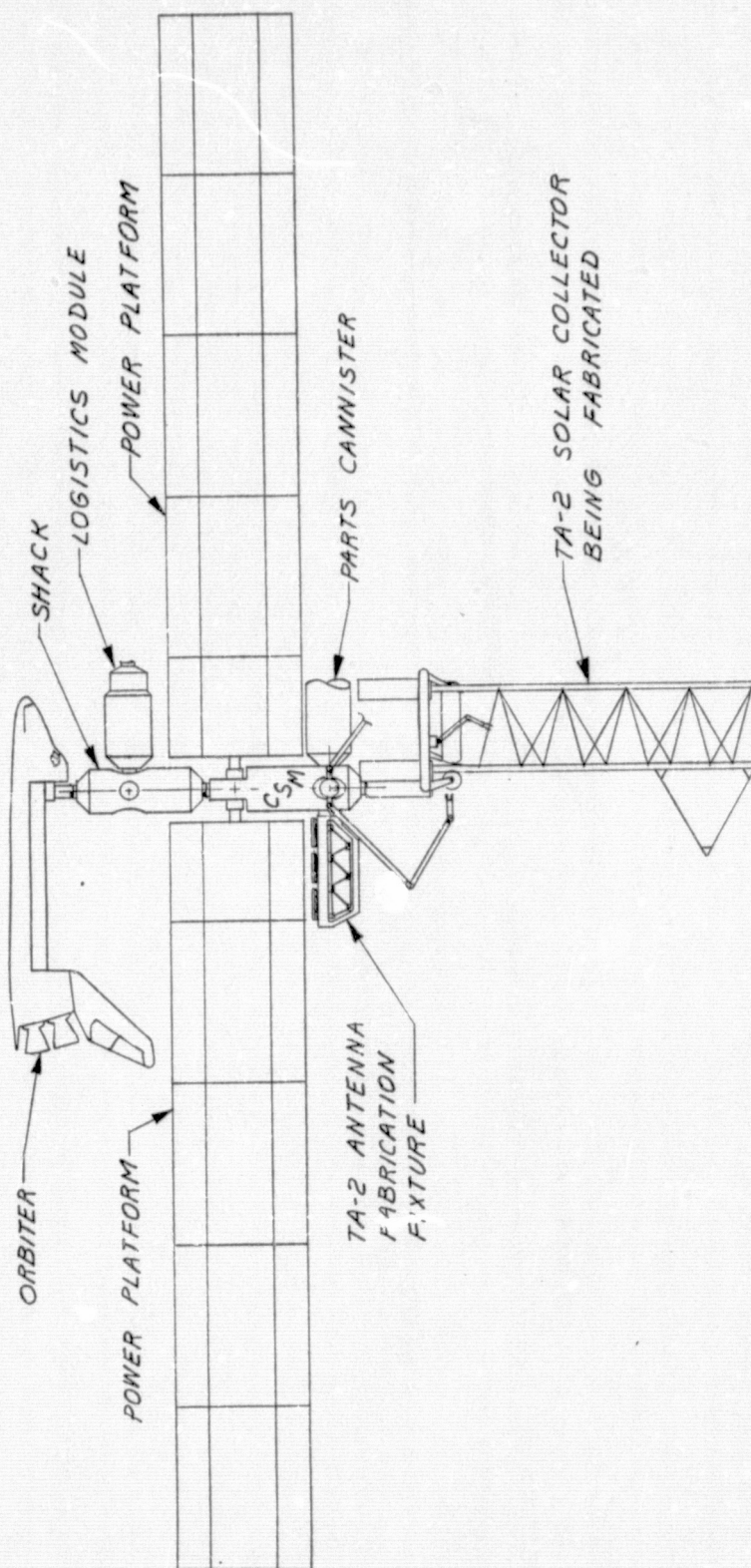
Figure 16-18, Sheet 2, shows a more elaborate base with the addition of a construction shack and logistics module. The power platform is shown rotated from the view of Figure 16-1. A more sophisticated construction project is shown - the manufacture of a TA-2 antenna and solar array in space.



CR60

16-29

Figure 16-18. (Shuttle Tended) — 150 kW Power Plant Balanced Double Gimbal (Deployed) $\eta = 2$ (Sheet 1 of 2)



16.7 FABRICATION/ASSEMBLY FACILITY

The in situ construction concept shown for the in-place construction (Figure 16-19) of a small (150 kW) array uses two wings on separate gimbals. Two fixtures along with a beam-fabricating module are transported in a single Orbiter launch. Each fixture is placed at the appropriate interface on the construction support module and the fabrication module is moved to the various beam positions to fabricate the array longerons. The array longerons are moved through the holding fixture as a pair and the solar cell blanket is unrolled from its drum until it is suspended entirely between ends of the array structure.

The ladder concept shown in Figure 16-20 is a flat planar structure consisting of four longeron beams and several transverse beams with solar cell blankets suspended in the open bays. The array is 28 meters wide and varies between 62 meters at 150 kW to 200 meters at 500 kW. The final size selected as appropriate is 109 meters long and 250 kW. The array structure is entirely composed of beams triangular in section, 1.0 meter on a side and made from graphite-epoxy or graphite-polyimide composite construction. A two-piece pallet for transporting the system and materials within the Orbiter is unfolded to form an assembly fixture and upon completion becomes the primary structural support for the array. A beam fabricating module is used at various locations on the fixture to fabricate the composite longerons.

The 10-meter beam prototype of Figure 16-21 shows that the same fabrication module and assembly fixture can be used to construct an array structure consisting of three longeron beams and diagonal truss tubes with a solar cell blanket cover on one side of the truss. The diagonal truss members would be prefabricated and installed via EVA operations. A separate gimbal mechanism and gimbal truss would be installed on one end of the truss.

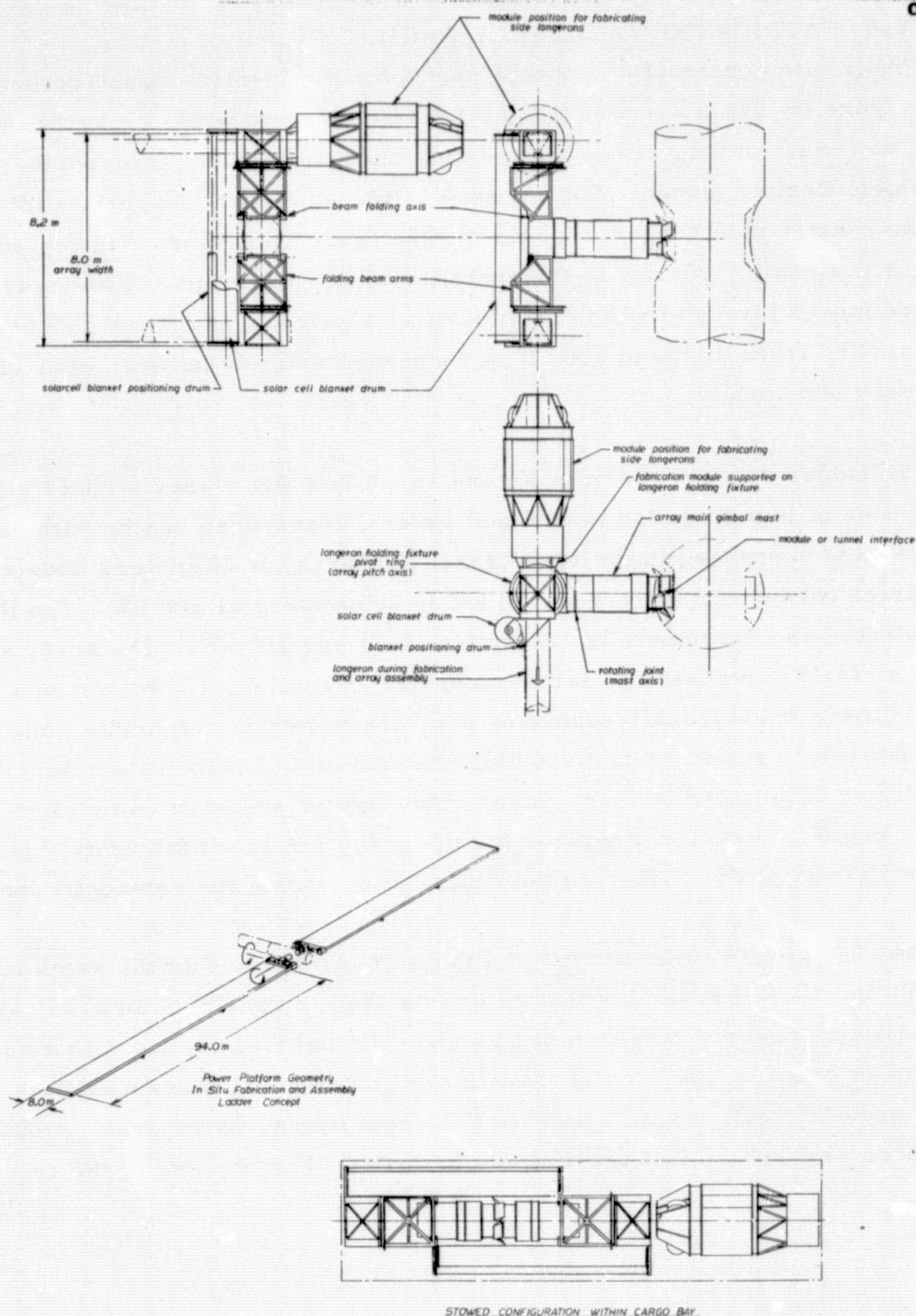


Figure 16-19. 150-kW Solar Array In Situ Construction - Fabrication/Assembly Facility

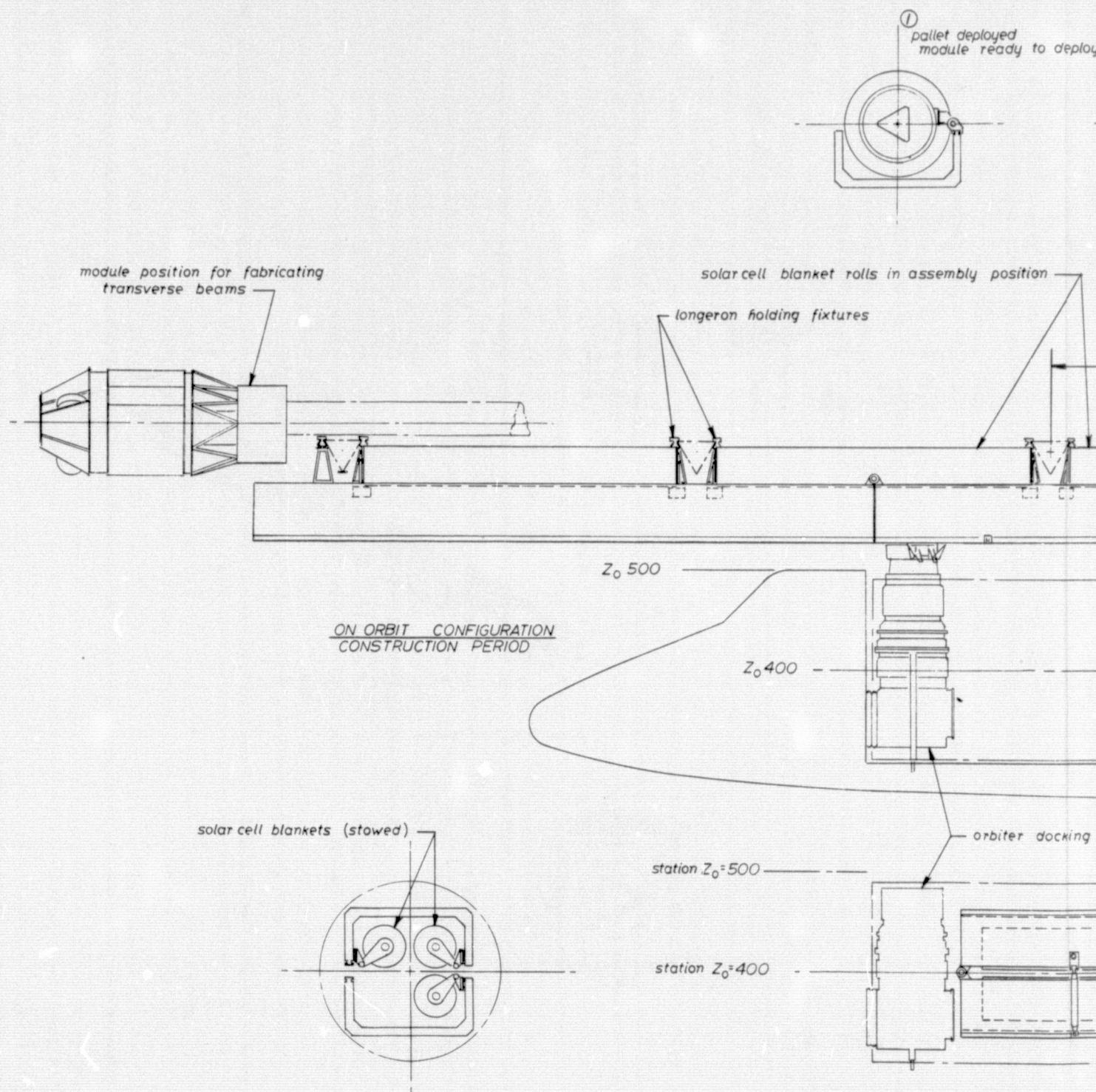
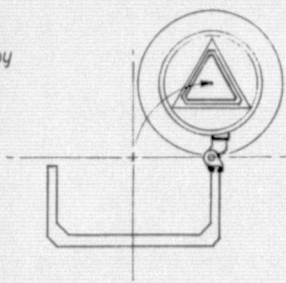
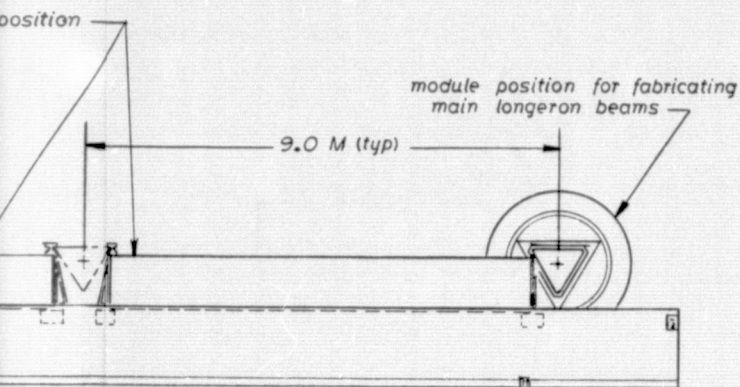


Figure 16-20. 150- to 500-kW Solar Array Ladder Concept – Fabrication/Assembly Facility

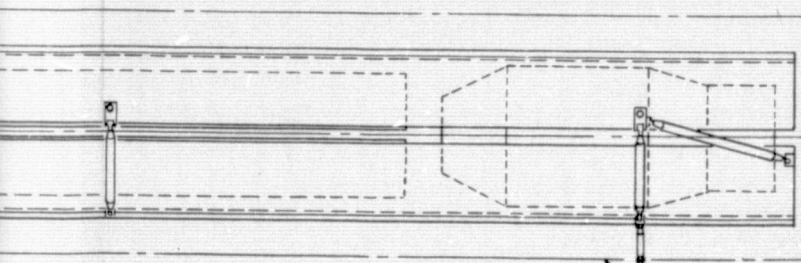
ready to deploy



- ② deploy module to operating positions ~
• this position for fabricating transverse beams

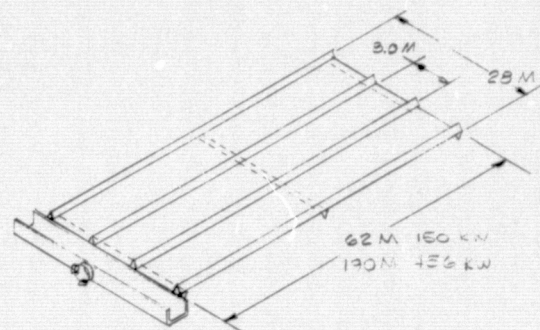
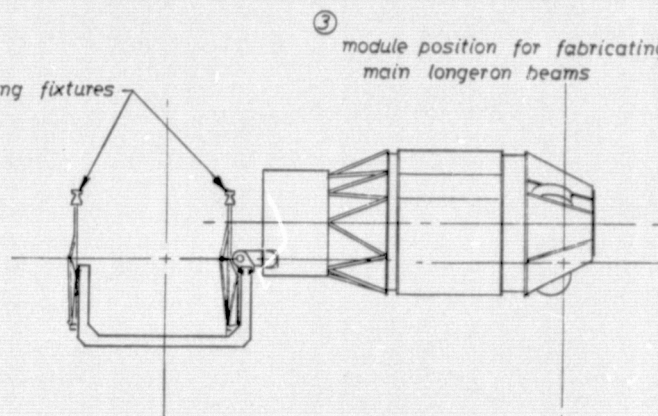


oriter docking module



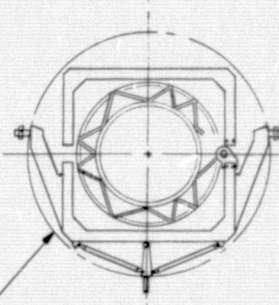
STOWED CONFIGURATION WITHIN CARGO BAY

longeron holding fixtures

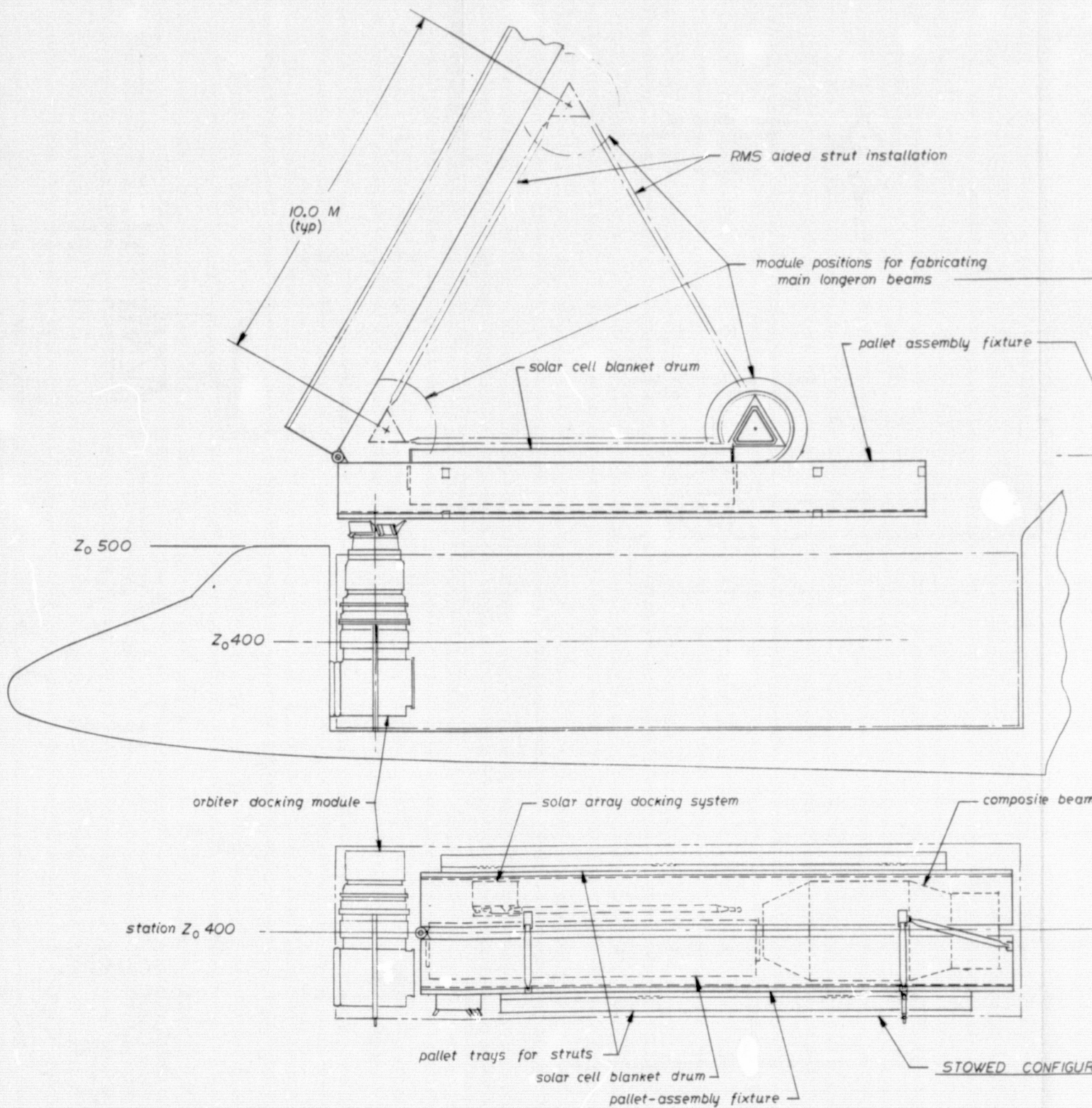


Power Platform Geometry
Ladder Concept

orbiter clearance line $Z_{o.427}$
payload hardpoints $Z_{o.415}$



FOLDOUT FRAME



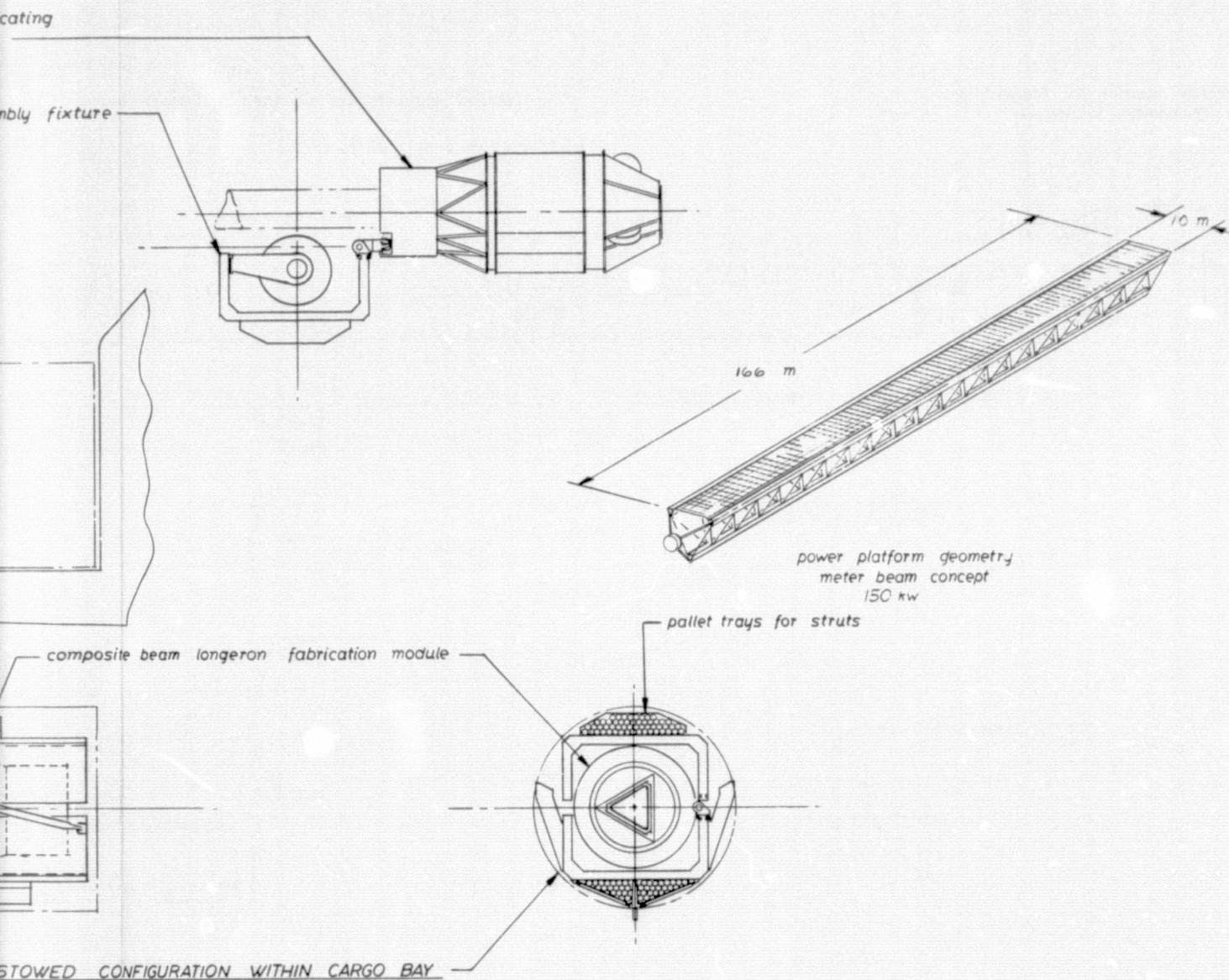


Figure 16-21. 150-kW Solar Array 10-Meter Beam Prototypical Concept — Fabrication/Assembly

16.8 BEAM AND TUBE MODULE

The 1-meter beam composite weave equipment is the module which fabricates the 1-meter triangular section beams which are prototypical of TA-2 longerons. This module, shown in Figure 16-22, consists of a structure with a centerline, triangularly shaped mandrel; reels for longitudinal corner tows; a rotating loom for laying composite tows in an open double helix on the mandrel; furnaces for curing the composite; a fixture for installing fittings for joining beam elements, and finally a mechanism for pulling the beam from the mandrel. Thermal energy carried out of the furnace by the fabricated beam is the largest makeup energy of the module which consumes a total of approximately 2.0 kW electrical power for a production rate of 0.1 meter per minute. For this reason, the module power requirement is very sensitive to production rate as is the length of the furnace necessary to provide the appropriate cure times.

The beam and tube making module (Figure 16-23) was designed for use in constructing the TA-1 antenna in space. The mechanical components of the tube maker could also be used to make tubes for the larger TA-2 antenna, and possibly the diagonal struts for the TA-2 solar array.

Because of the resin, the composite tube maker is in a pressurized compartment (Figure 16-24). This also allows shirtsleeve astronaut monitoring in the early stages of development.

Preimpregnated tow is wound circumferentially as well as longitudinally to give multidirectional strength and stiffness. The fabrication process is continuous which allows tubes of any length to be manufactured.

The other portion of the module (Figure 16-25) is devoted to assembling the triangular cross-section structure from three long tubes (longerons) and many short diagonal tubes. Fittings which are prefabricated on the ground are used to attach the diagonals to the longerons. Assembly is accomplished in a vacuum and the shell is primarily a sun shade and meteoroid shield. Assembly may be done manually by astronaut or automatically by robot. The feed speed of the longerons must be coordinated.

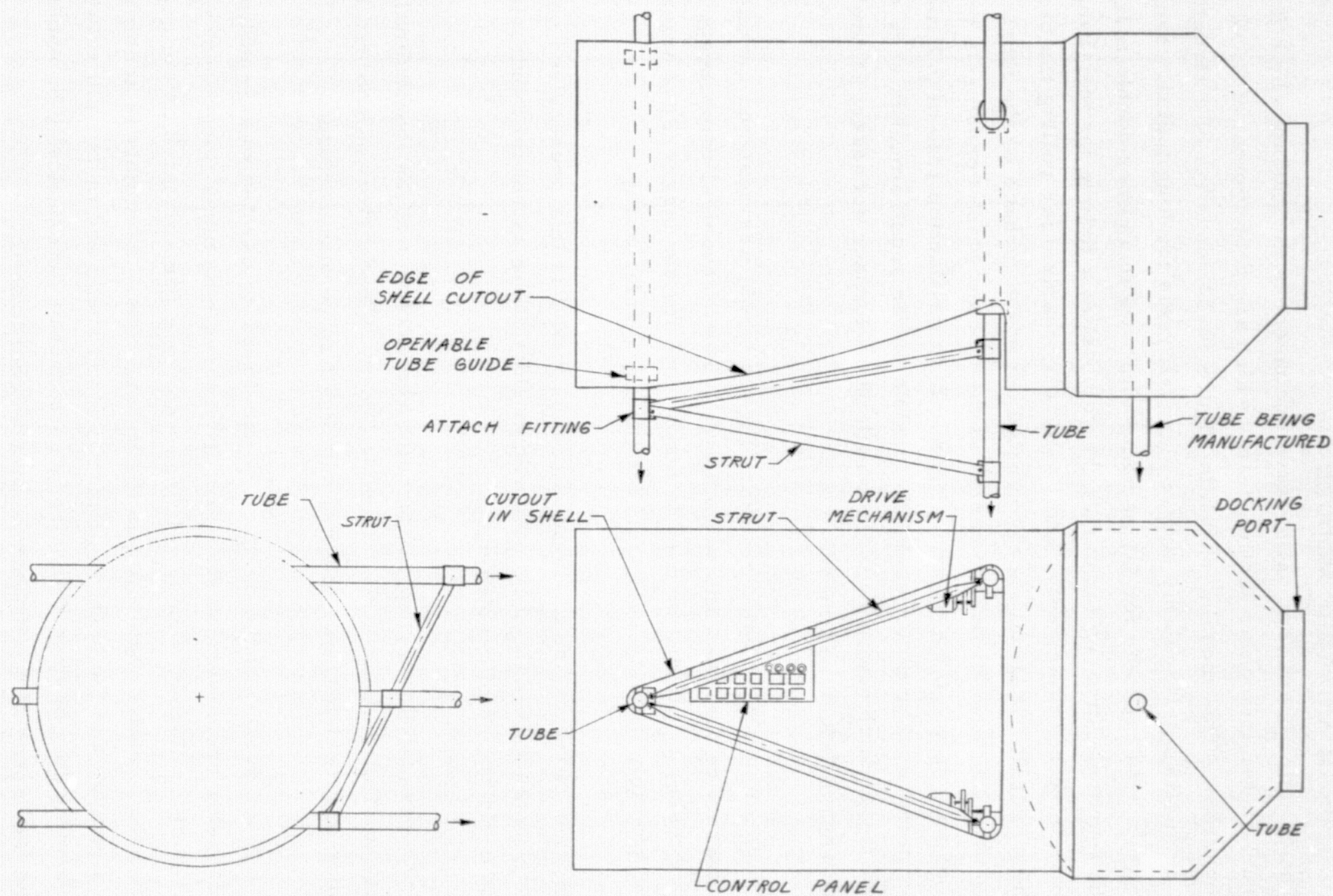


Figure 16-22. Beam and Tube Making Module

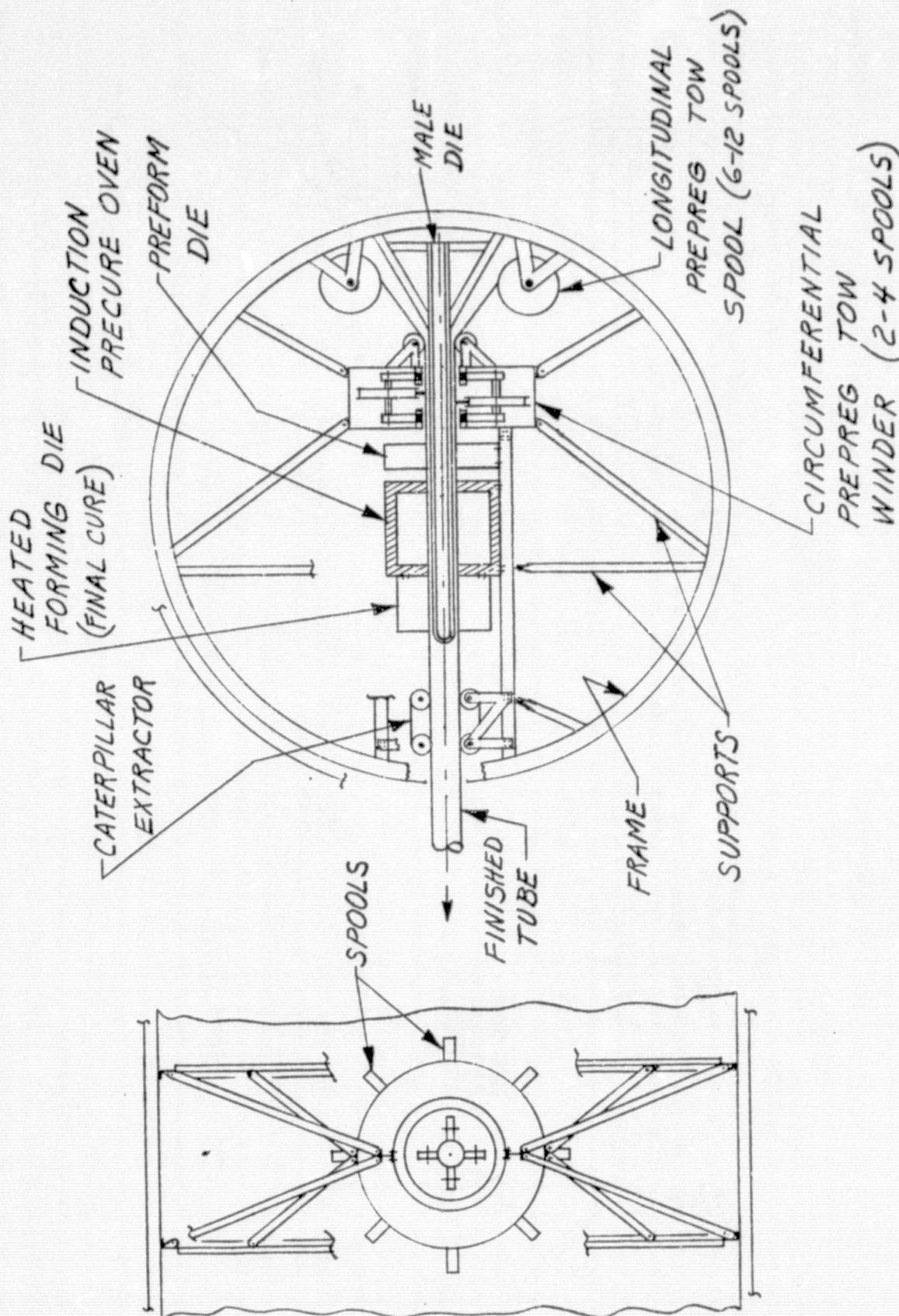


Figure 16-23. Composite Tube Making Machine

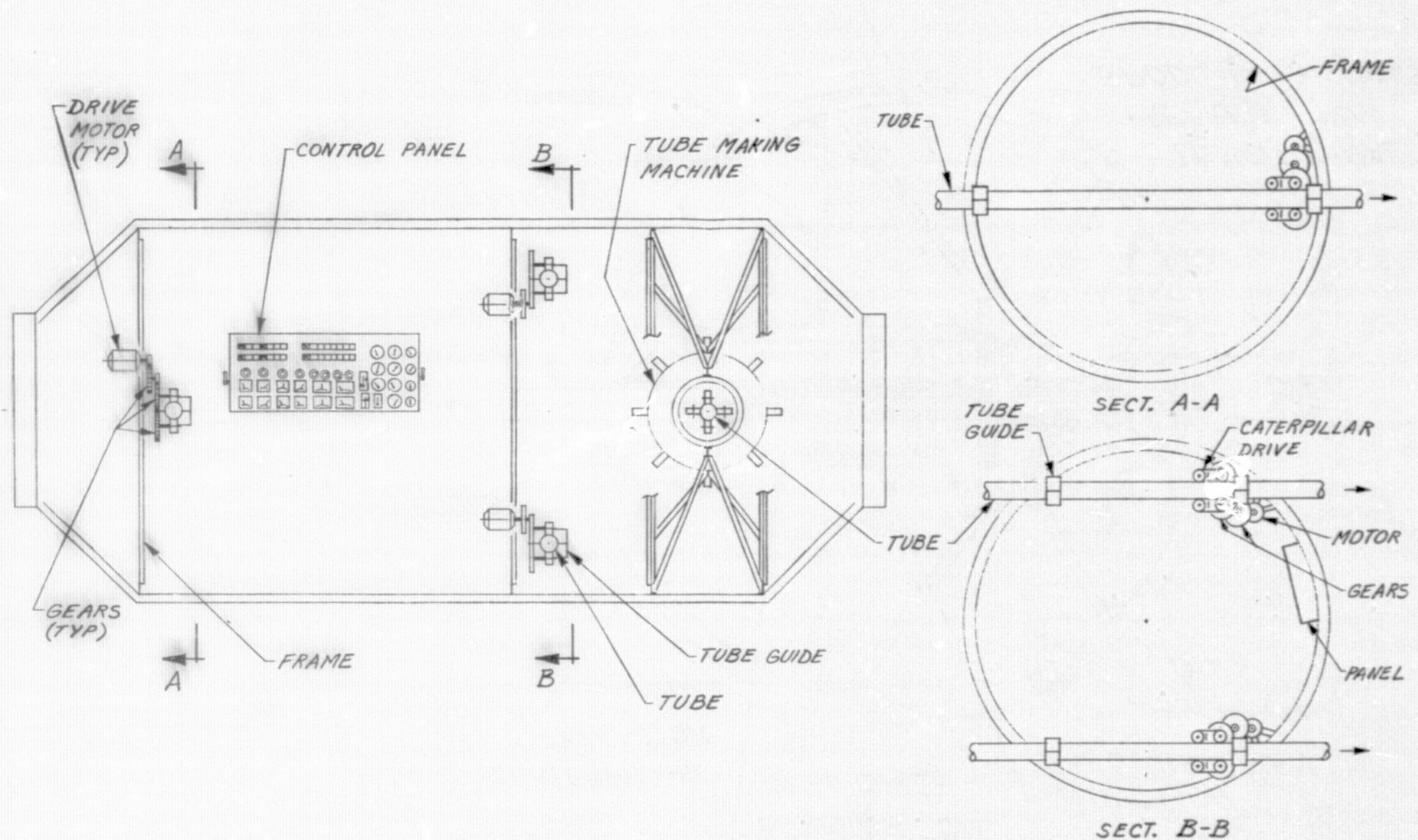


Figure 16-24. Inboard Profile - Beam and Tube Making Module

FOLDOUT FRAME

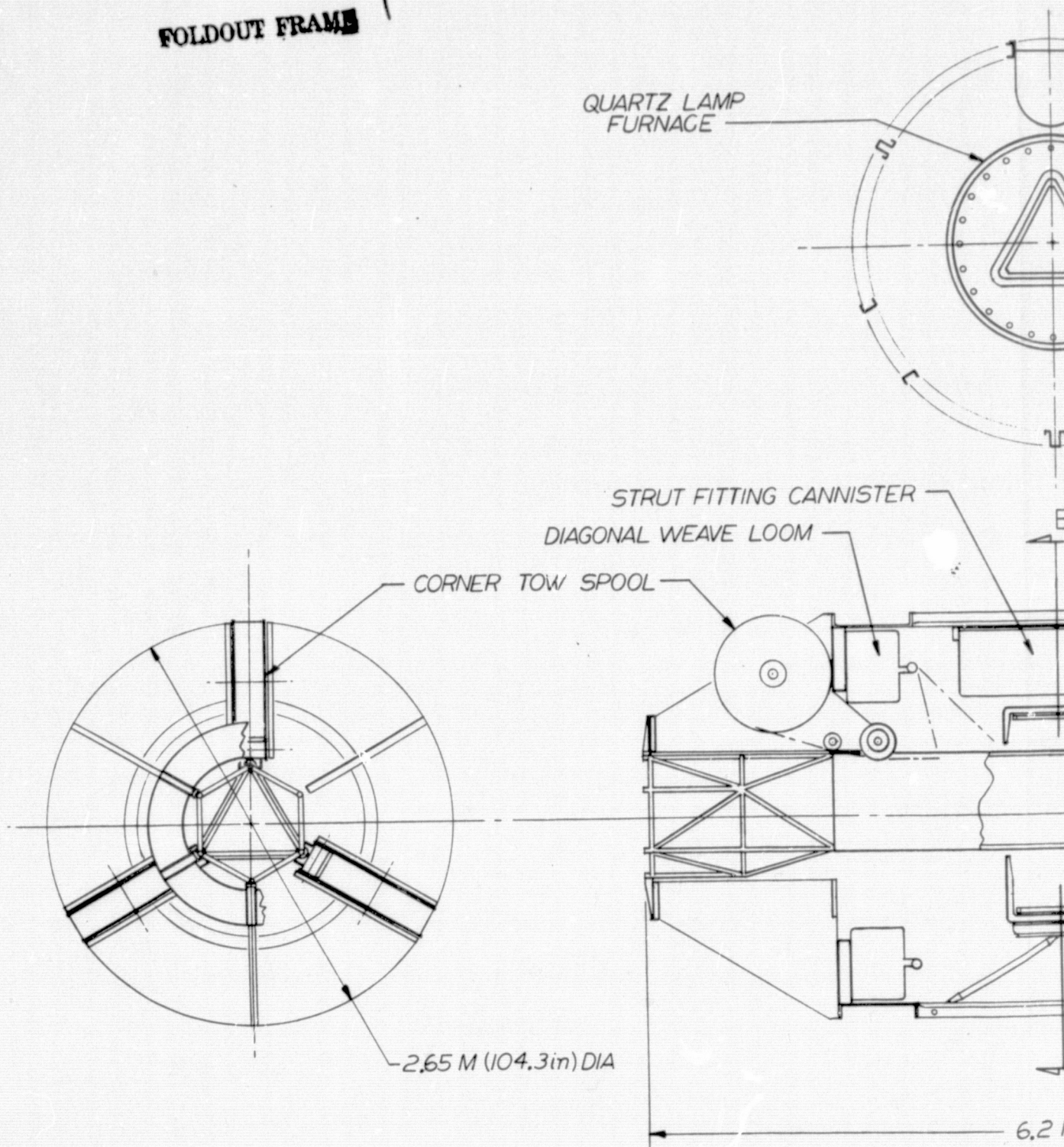
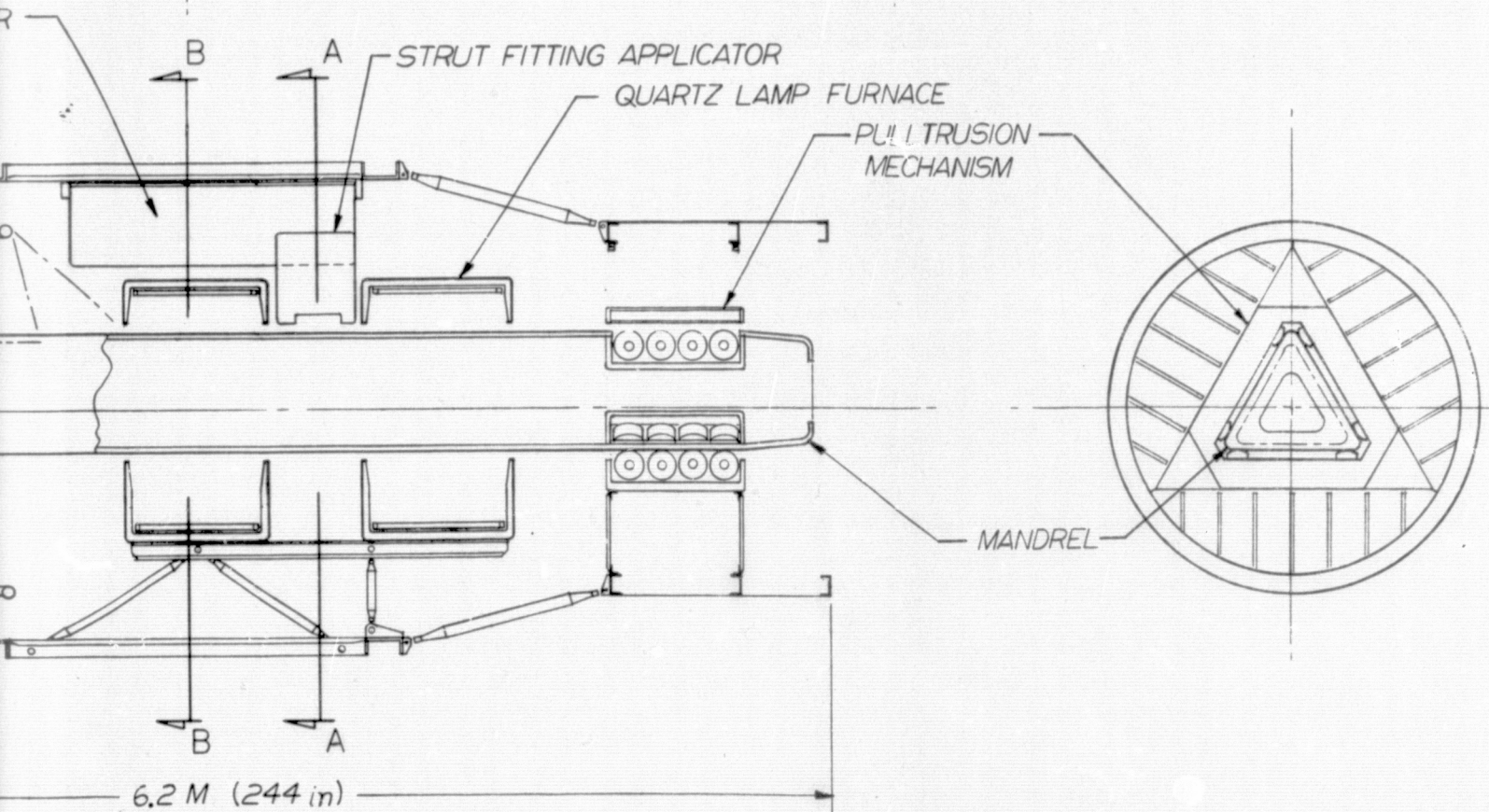
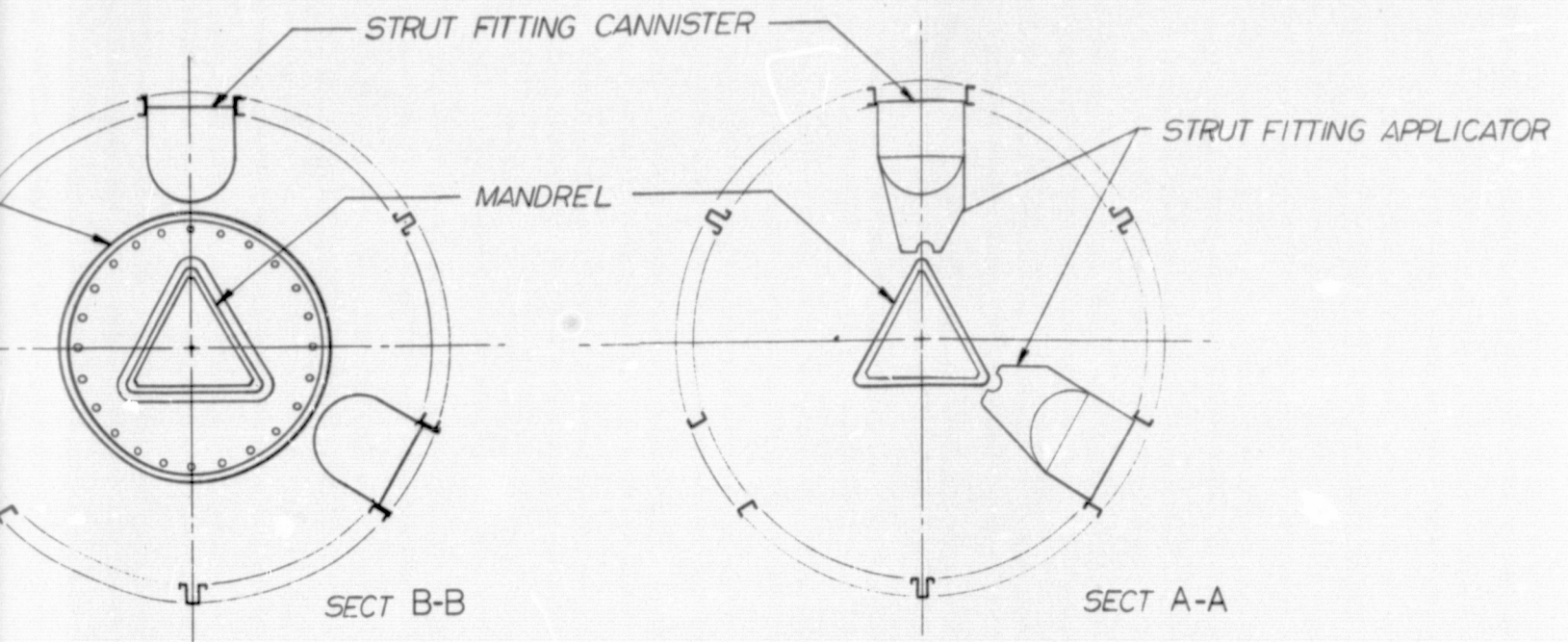


Figure 16-25. 1-Meter Beam Diagonal Weave Composite Concept — Fabrication



Section 17

SUBJECT REFERENCE MATRIX AND TABLE OF CONTENTS

17.1 SUBJECT REFERENCE MATRIX

The Subject Reference Matrix (Figure 17-1) provides a cross-reference between study tasks, and the particular volume of documentation where the most significant portion of the subject matter of the task is discussed. In several cases, detailed descriptions of a particular task may be found in an Appendix volume, whereas a synopsis of the effort will also appear in the Technical Volume, for example.

17.2 PART 1 AND 2 TABLE OF CONTENTS

Following Figure 17-1 is the Table of Contents of each volume of documentation generated during the study for Parts 1 and 2, and these may be used for reference to determine a more detailed breakdown of the subject matter in each volume. Tables 17-1 through 17-10 list these volume contents.

	PART 1				PART 2					PART 3						
	EXECUTIVE SUMMARY VOL. 1	TECHNICAL REPORT VOL. 2	APPENDICE ^a VOL. 3		EXECUTIVE SUMMARY VOL. 1	TECHNICAL REPORT VOL. 2	APPENDICES VOL. 3				EXECUTIVE SUMMARY VOL. 1	TECHNICAL REPORT VOL. 2	APPENDICES VOL. 3		SUPPORTING RESEARCH & TECHNOLOGY VOL. 4	COST & SCHEDULE
			OBJECTIVE DATA BK 1	OPTION DATA & COSTING BK 2			PROGRAM REQMT'S BK 1	SUPPORTING DATA BK 2	SUPPORTING DATA BK 2	COST & SCHEDULE BK 3			SUPPORTING DATA BK 1	SUPPORTING DATA BK 2		
PART 1 DEFINE AND EVALUATE PROGRAM OPTIONS																
TASK 1 OBJECTIVES																
1.1 REVIEW OF DATA BASE																
1.2 PREPARATION OF DATA SHEETS																
1.3 CORRELATION OF THEME ELEMENTS AND OBJECTIVES																
1.4 PRELIMINARY SCREENING OF OBJECTIVES																
TASK 2 ESTABLISHMENT OF MISSION DESCRIPTIONS																
2.1 PREPARATION OF MISSION SEQUENCES																
2.2 ANALYSIS OF LEVEL 3 REQUIREMENTS																
2.3 DEFINITION OF THEME ELEMENTS																
TASK 3 ESTABLISHMENT OF SPACE STATION PROGRAM OPTIONS																
3.1 ESTABLISHMENT OF GROUPS OF OBJECTIVES																
3.2 ANALYSIS OF CRITICAL LEVEL 4 REQUIREMENTS																
3.3 IDENTIFICATION OF SPACE STATION SYSTEM CONFIGURATIONS																
3.4 PRELIMINARY COST AND SCHEDULING OF SYSTEM OPTIONS																
3.5 IDENTIFICATION OF PROGRAM OPTIONS																
PART 2 DEFINE AND EVALUATE SYSTEM OPTIONS WITHIN SELECTED PROGRAM OPTIONS																
TASK 4 DEFINITION OF CONFIGURATIONS																
4.1 DEFINITION OF OBJECTIVE ELEMENTS																
4.2 SYNTHESIS OF DESIGN REQUIREMENTS																
4.3 DEFINITION OF SPACE CONST BASE AND MISSION HARDWARE ELEMENTS																
4.4 IDENTIFICATION AND ANALYSIS OF KEY DESIGN COST DRIVERS																
4.5 DEFINITION OF TRANSPORTATION SYSTEM ELEMENTS																
TASK 5 ANALYSIS OF OPERATIONS																
5.1 PREPARATION OF CRITICAL MISSION SEQUENCES																
5.2 DEVELOPMENT OF FUNCTIONAL/PERFORMANCE REQUIREMENTS																
5.3 ANALYSIS OF TRANSPORTATION SYSTEM																
TASK 6 SELECTION OF PRIMARY CONCEPTS																
6.1 DEVELOPMENT OF PROG REQMTS DOCUMENT																
6.2 DEVELOPMENT OF COMPARISON CRITERIA																
6.3 DEFINITION OF SYSTEM OPTIONS																
6.4 COMPARISON OF SYSTEM OPTIONS																
6.5 COST AND SCHEDULE DATA																
TASK 7 SYSTEM DESIGN																
7.1 SHUTTLE SYSTEMS CAPABILITY REVIEW																
7.2 MISSION HARDWARE REQMTS EXPANSION																
7.3 MISSION HARDWARE DESIGN																
7.4 CONSTRUCTION SYSTEM DESIGN																
7.5 SUPPORT SYSTEM DESIGN																
7.6 SCB SYSTEM OPTION SYNTHESIS																
TASK 8 REQUIREMENTS ANALYSIS																
8.1 MISSION HARDWARE REQMTS ANALYSIS																
8.2 SYSTEM REQMTS ANALYSIS																
TASK 9 PROGRAM DEVELOPMENT																
9.1 PROGRAMMATIC ANALYSIS																
9.2 PROJECT PLANNING																

Figure 17-1. Subject Reference Matrix

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PART 1 FINAL REPORT - EXECUTIVE SUMMARY VOL 1

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2.5	Space Processing	8
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SUPPORTING STUDIES

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- 7.2 Silicon Ribbon and Solar Cell
Blanket Production Plant

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Table 17-3

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CONTENTS

This book consists of the Space Systems Objectives Evaluation and 10 objectives data packages in the following order:

1. SATELLITE POWER SYSTEM
2. NUCLEAR ENERGY
3. EARTH SERVICES
4. SPACE COSMOLOGICAL RESEARCH AND DEVELOPMENT
5. SPACE PROCESSING
6. CLUSTER SUPPORT
7. DEPOT
8. MULTIDISCIPLINE SCIENCE LABORATORY
9. SENSOR DEVELOPMENT
10. LIVING AND WORKING IN SPACE

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